STUDY OF CONCEPTUAL DEEP SPACE MONITOR COMMUNICATIONS SYSTEMS USING A SINGLE EARTH SATELLITE

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1.0 OBJECTIVES, GROUND RULES AND SCOPE OF STUDY

Communication channels between spacecraft and the earth have in the past performed adequately with acceptable data rates and for the number of spacecraft requiring support. This study was undertaken to investigate situations in the 1975-1985 time period when larger data rates will be required and the number of spacecraft requiring communication channels will be increased. The stated objectives of the study were: (a) to determine the feasibility and capability of concepts for deep-space communications employing single earth satellites, (b) to perform parametric analyses to indicate the more promising integrated system concepts, and (c) to expose technical problem areas and to indicate how such problem areas can influence the choice and performance of candidate systems.

Given and evolved study ground rules were:

1. Orbit a single earth satellite as opposed to multiple synchronous satellites for communications relay between earth and a variety of planetary spacecraft.
2. Each satellite will be assigned full time to each spacecraft.
3. Operational time period - mid 70's to mid 80's.
4. Operational lifetime - comparable to spacecraft mission duration (maximum of 3 years).
5. Select satellite earth orbits to provide continuous line of sight between the satellite and spacecraft using orbit precession and no active plane changing.
6. Consider launch vehicle capabilities from the eastern test range and the western test range.
7. Launch vehicles considered: Saturn-IB/Centaur, Saturn-IB and Titan IIIC.
8. Frequencies of interest - 2 to 100 gigahertz
9. Satellite antenna gain - 60 to 80 db
10. Command power - minimum of 10 kW radiated.

The 9-month study resulted in a parametric analysis of deep space monitor communication systems consisting of a single earth satellite and associated interplanetary spacecraft. The accomplishment of the study objectives required parametric analyses involving considerations of frequency, bandwidth, radiated power, data rate, antenna size, weight and volume, and orbit inclination and altitude. Evaluation of support requirements were also necessary as were antenna fabrication and deployment, equipment modules, attitude control, power sources, micrometeoroid dust and radiation shielding, launch vehicles and site selection, and logistics and resupply. Considerations were given to system integration and tradeoffs as required to monitor both manned and unmanned spacecraft for planetary missions.

2.0 TECHNICAL APPROACH/STUDY PLAN

A flow diagram of the study plan is shown in Figure 2-1. Eight primary work areas are noted in the diagram with a chronological flow from left to right. Mission analysis was defined as the area deriving available launch vehicle payload capabilities and selection of appropriate launch vehicles. The difference in launch vehicle payload capabilities for the eastern and western tests ranges were also noted here. Orbit selection was investigated under Mission Analysis and included the line of sight to spacecraft trajectory analysis and the considerations of the selected orbit altitudes thus derived to orbital environment conditions. The line of sight to spacecraft trajectory analysis was one of the many unique and important problems of the study. Orbits were required which would assure continuous line of sight from the satellite to the spacecraft for a variety of missions in order to establish feasibility of the DSMCS concept.

The Mission Analysis material was then fed into three other work areas as noted in the figure. These areas were initially investigated separately. After substantial work was completed in these areas they were studied together and iterated. The Available Satellite Discipline Performance area consisted of a compilation of state of the art and a prediction of advances in state of the art into the Deep Space Monitor Communication Satellite (DSMCS) time period for all the required technological disciplines. The Satellite Discipline Requirements area presented desirable requirements for the DSMCS disciplines as dictated by the initial concept. Primary satellite technical discipline areas included: communications, power, attitude control, and environment control. Through the course of the study the desirable requirements were iterated with predictions of advances in discipline state of the art and tradeoff adjustments made. Satellite/Spacecraft Interface Considerations were also considered to a secondary degree and their influences were used for additional tradeoffs in the available discipline performance/discipline requirements iterative loop.

The iterative results of the last three work areas were fed into the Satellite Discipline Design Interface and the Satellite Conceptual Design areas. Prior to firming up the satellite conceptual designs, the interrelations of all satellite disciplines were investigated. Nine representative conceptual designs were presented with four preferred configurations. In retrospect, the Degree of Program Feasibility was discussed and Technological Problem Areas noted.

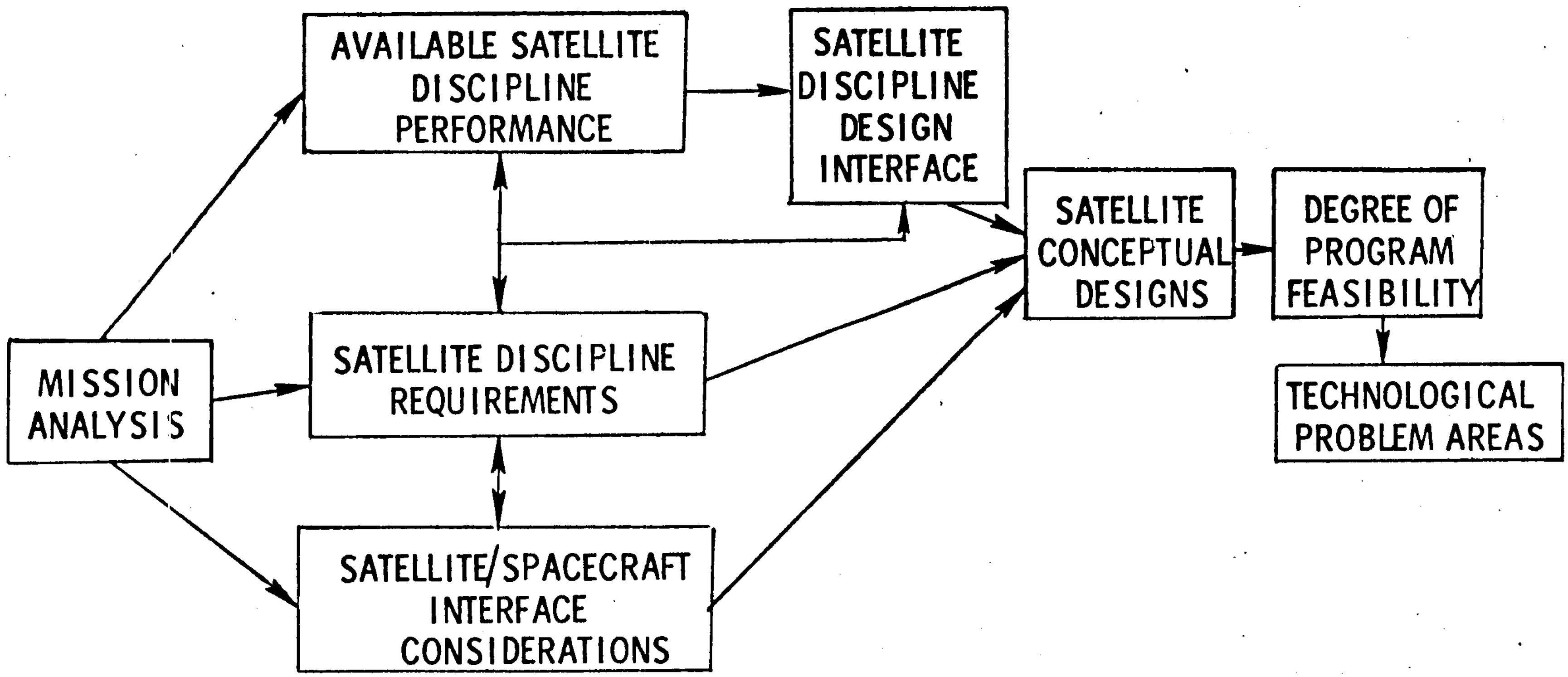

Figure 2-1. Study Plan and Work Areas

3.0 STUDY MILESTONES

At the beginning of the study a good portion of the technological and operational obstacles were easily recognizable. As the study progressed and more mature understanding developed, additional obstacles became apparent. Some of the hardware related milestones, while understood for study purposes, required capabilities beyond their present state of the art and are noted later as technological problem areas. Study obstacles requiring understanding to a degree that the study could not progress properly without a firm fix on them were termed milestones. A list of the milestones encountered follows:

- Line of sight to spacecraft through non-occulting orbit selection.
- Environment limited cooled amplifiers either parametric or maser.
- Closed loop cryogenics for amplifier cooling
- 2-100 gigahertz cryogenic amplifiers
- 2-100 gigahertz efficient transmitters
- High power transmitter thermal control
- Antenna electrical and mechanical design
- Antenna electrical to mechanical relationships
- Antenna thermodynamic analysis
- Manufacturing and deployment characteristics for one piece, petal, and inflatable antennas
- Feed position erection and operation tolerance
- Extrapolating nuclear power system technology
- Nuclear power system thermal control
- Establishing solar panel power system interface with attitude control system requirements
- Establishing attitude control system performance related disturbance torques
- Attaining required attitude control system for the required pointing accuracy

4.0 BASIC COMMUNICATION SYSTEM RELATIONSHIPS

The functional design of the DSMCS is bounded by the limitations of the communication system components as well as the components of auxiliary peripheral systems. A tentative system configuration is formulated here which meets the DSMCS mission requirements and is compatible with the probable characteristics of connecting systems. The communication system design configuration, although it is not optimum in any particular sense does demonstrate total feasibility, expose critical areas and indicates alternative approaches. The intention here is not only to show how it can be done but what is the rationale leading to the recommended design. The description of the proposed communication system configuration is preceded by an itemization of the objectives and assumptions of the design. The operations including track, calibrate and acquire are considered including detailed itemization of the key equipment characteristics including reliability.

4.1 GOALS AND CONSTRAINTS

The primary goal of the DSMCS is to monitor spacecraft operating in deep space. The receiver monitoring the spacecraft must be as sensitive as practically possible in order to conserve spacecraft energy expended per bit of information transferred. Implicit in a maximum bit rate transfer per energy expended criteria is a need for a high gain antenna which tracks the spacecraft continuously and a coherent receiver. DSMCS or some combination of DSMCS and an earth-bound receiver and processor must be capable of extracting doppler information for spacecraft trajectory computation. The beam position relative to a known coordinate system is also necessary for accurate trajectory tracking. The long mission times suggest the need for a ranging capability in the receiver in order to compensate for the large tracking errors that build up due to oscillator drift. Another DSMCS goal is a command capability. The spacecraft command, of course, would be formulated at a ground station and relayed to the DSMCS.

The general policy at each functional level was that of minimizing operating complexity within DSMCS. This is by no means easy since there are numerous control loops whose characteristics are adaptable to specific situations which must be locked quickly in the proper order and kept tracking the received signal. It will be seen that limiting the complexity reduces system weight and more importantly increases system reliability. In the direction of reducing development time and enhancing reliability, where possible, the techniques and system components used are those which have been utilized on the unmanned space network and which will soon be implemented in the manned Apollo communication network. The one constraint on the down-link or DSMCS to earth-bound station channel is that it be compatible with the ground instrumentation of the unified S-band system. Although many of the 30-foot dishes of the manned spaceflight network would be available in the late 70's, it appears to be an unnecessary complexity to continuously transfer deep spacecraft receptions in real time to the ground. This will be particularly true for missions in the late 70's and 80's, where the transmission times will be great and the bit rates relatively low. For this reason, it was assumed that once the DSMCS antenna is oriented and tracking the spacecraft, the monitored information will be stored and returned to the ground station at a higher rate periodically. A single polar ground station would suffice for most DSMCS orbits.

4.2 SYSTEM DESIGN

In order to reduce the communication system complexity, increase flexibility and to improve reliability, it is necessary that as many control functions as possible be located on the ground. The majority of the controls entail adjusting loop oscillators, sweep rates and/or loop characteristics for minimum acquisition time and improving general performance. These controls are best adjusted by large capacity data processing units which are capable of holding and utilizing the various system interface functional relationships and mission history necessary for true optimum control. Thus, whenever possible receiver loops should be terminated at the ground station where large computers are available. A ground control processor has the additional advantage of almost unlimited flexibility to adapt to unexpected situations. Figure 4.2-1 shows a block diagram of the DSMCS communication system where five loops are closed in the satellite. These closures are considered unavoidable in order to effectively meet DSMCS goals. An additional rf loop and both antenna tracking axes loops, and a range stream loop are closed at the ground station in either delayed or real time. When necessary, the deep space information is recorded for burst mode transfer by a unified S-band channel to the ground.

A detailed description of the system will begin with the transmitter group at the top of the diagram. A very stable frequency standard supplies the frequency reference for appropriate multiplication and division in the transmitter synthesizer. The synthesizer VCO (voltage control oscillator) is locked to either the reference or a programmed acquisition voltage. The VCO output when locked must be of high spectral purity in order that useful doppler measurements might be made through the spacecraft loop. The VCO output is multiplied up to the transmit frequency and modulated by the spacecraft commands and range code. The range stream, if implemented, will be infrequently used if a sufficiently stable frequency standard is used since doppler tracking can fill in between ranging fixes. Following modulation, the signal is filtered and amplified for transmission. The amplified signal may be directed to either a smaller acquisition antenna or the tracking antenna or a load for test purposes. A diplexer would precede signal transmission if real time coherent doppler tracking is incorporated. A polarization control block has been included although it may be necessary for most missions. Three-axes stabilization will undoubtedly be maintained at the spacecraft and DSMCS, and therefore the only reason for polarization control would be to discriminate between two channels of data in orthogonal planes. The antenna feed must also provide error signals for tracking in the orthogonal planes.

The received deep space sum channel signal comes from either the main tracking or acquisition antenna diplexer into the antenna selector. The receive selector also connects the receiver to controlled absolute power levels for sensing low noise amplifier performance. The cryogenic amplifier raises the signal and noise sufficiently high so that the noise of subsequent stages will not affect the signal-to-noise ratio. The amplified signal is then divided and mixed at two separate points. One mixer utilizes an incoherent local oscillator derived from the receiver synthesizer. The desired mixer products are amplified and then multiplexed into a convenient format for transmission or storage and eventual transmission to the ground station over a unified S-band communication link. A sample of the receiver synthesizer is also stored coincidentally with the data and transmitted via the S-band link so that doppler tracking of the deep space signal may be achieved at the ground processor in spite of storage medium distortions. The other cryogenic amplifier output is mixed with a coherent oscillator. The coherent oscillator permits extraction of the phase and amplitude of the carrier in order to establish an AGC voltage and error signal in the antenna tracking channel. The sweep and phase control of the VCO and the loop characteristics will be either commanded from the ground station or controlled by a DSMCS internal program with certain program constraints commanded from the ground station. The rf carrier loop which is closed at the ground station is not intended to be a back-up for the DSMCS rf loop. The ground station loop will be controlled in real time or by a large capacity data processor and thus more useful data from the deep space transmissions as well as general DSMCS systems diagnostics can be more efficiently extracted. The DSMCS rf loop is intended as a convenient method of obtaining an AGC voltage error signals from low data rate telemetry. Another method of deriving AGC and angular error signals at the expense of carrier power is to increase the carrier power and modulate the spacecraft data in such a fashion that a large span exists between the lowest information spectrum and the carrier. By this method a simple amplitude detector could be used to control the AGC and angular error channel. The angular error signals which describe the position of the DSMCS beam relative to the spacecraft are digitized and commutated with other low data rate signals within the DSMCS system. The large dish antenna signals are utilized at DSMCS for attitude control error correction. The DSMCS frame orientation relative to the celestial sphere is derived from a star tracker. The orientation of the antenna beam relative to the DSMCS frame will be set before launch or will be calculated in flight. In addition to spacecraft angular error data and star tracker data, the range code and range correlation signal is provided to the digital commutator.

The DSMCS to ground transmission can be tailored to whatever format exists for unified S-band communications at the time of DSMCS launch with the usual telemetry spectrum replaced by the high data rate deep space information and DSMCS monitor signals. Recorded deep space data is taken from storage at approximately 20 times the record rate. The transmitter power is less than 5 watts for a 200 kilobit per second rate with an omnidirectional antenna on the satellite and a 30 foot diameter unified S-band antenna and receiver on the ground. The usual spacecraft closed loop tracking receiver must be implemented at the satellite so that the DSMCS location accuracy errors are sufficiently small so as not to affect the spacecraft tracking.

4.3 OPERATION

After having reviewed the general functional operation of the DSMCS communication system, a closer look at specific operations performed by the system and the critical components necessary to perform these operations is in order. Normal tracking operations will be reviewed first followed by calibration and acquisition operations. Qualitative evaluation of accuracy and stability of important measures are not possible until more specific details regarding operating parameters have been determined. Much useful system design, however, can be accomplished without this data.

4.4 NORMAL MODE

In a normal mode of operation the DSMCS aperture will track the spacecraft continuously. Not only must the receiving aperture be oriented toward the spacecraft but the orbit of the DSMCS must precess in order that the earth or lower atmosphere will not degrade the spacecraft signal transmission. The appropriate turning rate necessary to compensate for the difference between the spacecraft line of sight vector and the DSMCS orbit normal vector is at most, less than .6 deg/day. The required angular rate necessary to follow the spacecraft during each individual orbit is a function of the range and beamwidth of the antenna. If the small acquisition antenna is no less than one-tenth the diameter of the large antenna it would be used to track over the first 10% of the mission range. And if it can be assumed, the shortest maximum range of a deep space mission is 1.25 AU, then .125 AU would be the shortest tracking range for the large aperture. Figure 4.4-1 shows the diameter of the aperture as a function of operating frequency necessary to achieve the 3-decibel antenna beamwidth which would include the deep space spacecraft over the whole DSMCS orbit without correction. The angular beamwidth in aperture diameter in wavelengths is also shown. The line corresponding to .125 AU is the nominal state-of-the-art for solid parabolic reflector antennas in the study frequency range. Because of the similarity of the antenna state-of-the-art and the tracking limits, the antenna design upper limit will be 1.5×10^3 wavelengths and, therefore, tracking around the DSMCS orbit will not be necessary.

Not only must the DSMCS aperture be directed toward the spacecraft but its position must be known with respect to some reference system. The celestial tracker block forms a reference frame. The stability of this platform should be an order of magnitude better than the tracking antenna beamwidth. The angle cosines between the celestial platform and the attitude controlled tracking dish will be transmitted to the ground station for trajectory computations. The plots for .0625 and .125 AU of Figure 4.4-1 represent an upper limit of antenna beamwidth. The proposed Orbiting Astronomical Observatory celestial tracker maintains a $.016^\circ$ stability and therefore would be marginal for the .125 AU system and adequate with a factor of 4 improvement. The Orbiting Astronomical Observatory tracker is selected here to demonstrate the feasibility of this particular component and is not specifically recommended for this use. The mean time between failure of this particular platform is sufficient for DSMCS reliability goals. The error signals necessary to drive the large antenna may be generated by physically moving the beam or by the use of a monopole feed system. The beam motion, although it could be implemented by a moving feed is quite difficult because of reliability considerations. The feed system for an antenna is not easily interchangeable with a redundant unit and thus its reliability must be extremely high. A double monopole feed system intended to generate error signals causes a reduction in system sensitivity due to the insertion loss of the feed (.3 decibels). This represents approximately 20 degrees Kelvin noise temperature contribution for a 300°K thermometric temperature feed. In a normal mode the error signals are derived from the carrier and transmitted to the attitude control system in real time and to the ground processor in delayed time. The antenna tracking loop is closed at the DSMCS in order to provide quick response to environmental perturbations. The antenna loop is monitored at the ground to provide optimum control of the satellite antenna control loop characteristics.

Even if an rf ranging capability is implemented, a doppler tracking requirement will be necessary to resolve range within each range bit and as a backup for the wideband range. In order to achieve accurate doppler tracking, a very stable frequency standard must be provided. If it were not for the data storage arrangement, a time standard might be placed at the ground processor. A distortion induced on the stored data by the tape drive, which is locked to the frequency standard to reduce the very low frequency distortions, requires that a very stable timing pulse be recorded with the data. As indicated earlier the frequency standard provides a time base for the transmitter synthesizer and receiver synthesizer. In order to achieve the best doppler accuracy that is possible with present day phase lock-loops, a stability comparable to that of rubidium is necessary. Operational unmanned, and the in development manned, S-band doppler systems use a rubidium standard. The stability of two parts in 10^{11} from one second to twenty minutes produce velocity accuracies from .03 meter per second to .003 meter per second for closed loop operation. The accuracy is based on smoothing periods of one minute to five hours. The extrapolation of this technology from the existing ground systems to DSMCS is not as hazardous as it might appear since the break in the doppler loop represented by the DSMCS recorder is nearly identical to the break in the unmanned ground space network doppler loops where the transmit station with a frequency standard is not the same as the received station. In this case, microwave links transmit the data or carrier for coherent demodulation. Although the acquisition problems are difficult, the design of the transmitter, phase-lock loop and synthesizer is within the state-of-the-art. The frequency standard design is developmental. Prior to the discovery of the Mossbauer effect several models of 10^{11} stability clocks for space flight were being developed. At the time of the discovery one of the development efforts had nearly finished an ammonia clock which weighed less than 20 kg. A much smaller clock made of highly stable quartz crystal with a 10^{10} stability per day on a long-term basis has been developed for a geophysical satellite. Both of these efforts indicate that a flight model of 10^{11} short-term stability oscillator could be available for DSMCS.

It will be shown in the thermal control section that energy and command power limits of the DSMCS preclude continuous two-way doppler tracking. The 5 minute per hour command signal prevents DSMCS from closed loop tracking for any mission greater than .3 AU from the earth due to transmission time alone. Although one-way tracking is not as accurate, a clock similar to that recommended for DSMCS at the spacecraft would permit accuracies approaching that of two-way systems. An alternate approach would include a less sophisticated clock at the spacecraft which would accept an ultra stable carrier from DSMCS for generation of oscillation draft corrections. A third method would include a periodic command carrier sent from a ground based system which includes very stable carrier and which is then tracked through the spacecraft by DSMCS.

During the normal mode of operation, the receive spacecraft telemetry will be periodically transmitted to the ground processing unit. Since the total transient time and the reaction time for most deep space missions is comparable to the orbit period, a recorder playback once per orbit is probably sufficient. The number and location of stations necessary to permit playback only once per orbit is a function of the DSMCS orbit inclination and altitude and the ground antenna receive sector. Figure 4.4-2 shows the orbit of a satellite with an 80° inclination. The inside circle near the North Pole represents coverage of a receive station at Fairbanks, Alaska, for a minimum 10° elevation angle and a 1,000 kilometer orbital altitude. It can be seen that for this nominal configuration 10 of 14 orbits can be seen from 5 to 10 minutes at the station (10° elevation, horizon to zenith time is 7.3 minutes). A station located in Northern Europe could pick up the remaining orbits. The thirty-foot dish and receivers planned for the unified S-band system are more than sufficiently sensitive for the DSMCS to ground communication link with a low power transmitter. Once a connection has been established between the ground stations and a large-capacity data processor, the use of that ground station or stations could be time shared with other DSMCS type systems.

The spacecraft to DSMCS communication link and the ground to DSMCS communication link each require a storage medium for receive information. The ground station to DSMCS link storage would retain commands for the DSMCS and the spacecraft being tracked for eventual command of spacecraft at the appropriate time. The capacity requirements of this storage medium are modest but the access time must be short. Thus, a solid state memory with the additional high reliability would be in order here. A typical example is the Orbiting Astronomical Observatory memory which includes a small 200-kilobit magnetic core memory. The storage medium for the spacecraft telemetry will by necessity be magnetic tape assuming no large advances in the present technology of storage mediums. The recorder here is nothing more than a sequential copy machine of a large capacity with no access requirements. DSMCS diagnostic data for evaluation of its performance and spacecraft telemetry data will be multiplexed with the spacecraft telemetry data. The required capacity of the recorder lies between 7×10^6 bits and 1.4×10^8 bits. The extremes are represented by a 1×10^3 cycles per second spacecraft channel rate and a single orbit retention and 10×10^3 cycles per second spacecraft channel rate with a two-orbit retention. Endless loop recorders have been operated under similar environment requirements as that of the DSMCS at either end of the storage capacity spectrum. The Mars Mariner recorder at a 5×10^6 bit capacity was on 300 feet of magnetic tape and the Nimbus recorder is capable of storing 10^7 bits on 1200 feet of magnetic tape. In the case of the Mariner recorder the reliability of the total system was not significantly increased by the recorder alone. The Nimbus recorder did have a redundant unit in order to improve reliability. The need for a redundant recorder or any other critical DSMCS component will largely be dictated by subjective criteria on the importance of certain system functions and also on the availability of particular units.

The DSMCS to ground link will be compatible with the unified S-band system. A normal mode channel from the ground to the DSMCS will include the commands for the DSMCS and spacecraft. The down channel will include the spacecraft telemetry and DSMCS diagnostics in addition to verification of commands transmitted and their execution. In order to obtain meaningful spacecraft doppler and range data, the S-band link includes an rf loop and a range loop. The present two way doppler and range accuracy of approximately .003 meter per second and $\pm$ 15 meters respectively can be achieved with little power and are sufficient for extraction of the spacecraft doppler and range. The spacecraft rf and range loop will be established at the ground station by initially taking out the doppler induced by the S-band link followed by a demodulation of a local oscillator which should be in synchronism with the periodic pulse trains telemetered with the spacecraft data from the DSMCS stable oscillator. The acquisition time of the unified S-band link is less than 1 minute for an established near earth orbit.

4.5 CALIBRATION

A considerable number of tests must be performed on a spacecraft monitor station located on the earth before it will accurately track a spacecraft. The calibration test for a DSMCS in orbit is, of course, more difficult. Among the calibration tests necessary are the usual subsystem component tests of each unit ultimately leading to a test of the entire system. Of particular importance for the DSMCS calibration is the antenna performance test following deployment or simply orientation. A boresight error and gain must be established. It is not clear that an inflatable or rigid antenna will serve as the tracking antenna. This will be determined in the system optimization. In either case the unified S-band link will be established first to aid in estimating the DSMCS trajectory. Second, having determined the trajectory, the remaining components of the DSMCS will be tested and calibrated according to the command message given when the DSMCS passes over a ground station. The components will be checked out successively beginning with checks on the cryogenic amplifier sensitivity and command storage and logic. It will be assumed that a smaller acquisition dish, which is rigid, be available with known antenna characteristics. The other components will be tested through the simple horn antenna at the normal system frequency preferably to a ground station. By this method the transmitter and frequency standard may be checked for stability by comparison with the unified S-band link. Similarly, the bias and distortion induced by both telemetry recorders is determined by comparison to the rf phase-lock loop. Following calibration of the DSMCS subsystems, evaluation of the antenna is accomplished. The acquisition antenna boresight bias, with respect to the stabilized platform, can be obtained by tracking selected sources near the ecliptic such as the planet Venus or Mars. The primary antenna will then be deployed if it is not already. Experience with the particular material of which the antenna is composed will dictate the level of calibration necessary. However, it is probable that antenna patterns will have to be taken. The sun is too large in angular width for pattern determination, and the planets are not strong enough to permit an accurate determination of sidelobe characteristics. A strong ground transmitter could be used to make the pattern cuts. Cuts in both planes with both sum and error channels would be processed on the ground to generate bias errors and pattern functions, in order that the antenna servo loop characteristics may be controlled in an optimum fashion. A final check of the total system would be in tracking and ranging on a spacecraft while simultaneously tracking with a ground system. A weight penalty of two transmit frequencies on such a spacecraft would not be great if the DSMCS frequency were selected as a multiple of the unified S-band signal so that most components could serve dual duty.

4.6 ACQUISITION

The greater operational uncertainty in placing most loop control points at the ground station near the large data processors occurs during acquisition. It is very improbable that the acquisition by the DSMCS of a spacecraft could be accomplished during a single pass over one ground station. Thus, acquisition would require several orbits to complete and verify each step of the acquisition process. This is not too significant a penalty considering that acquisition should only be necessary once for every spacecraft mission track. At the outset of acquisition it is assumed that: the DSMCS has been calibrated, the DSMCS orbit has been accurately established, and the S-band link has been refined to the level that the affect of the DSMCS to earth range rate can be efficiently eliminated and that the DSMCS stable oscillator pulse train can be effectively stripped from the unified S-band telemetry so that time distortions due to the magnetic storage medium are eliminated. Also it will be assumed that command power during initial acquisitions is radiated from a ground station because of the DSMCS average power limit. Acquisitions from DSMCS command would take much longer.

The acquisition procedure for the short range portion of the spacecraft flight using the acquisition antenna would be essentially the same as that of the long-range section using the long-range dish if it is necessary. In both instances the initial spacecraft illumination by the ground station is achieved by directing the ground aperture and the DSMCS aperture towards the spacecraft trajectory predicted by earlier tracking instruments. The transmit oscillator is swept periodically during this illumination so that the carrier will definitely fall within the spacecraft loop bandwidth. Once spacecraft lock is achieved the spacecraft transmitter responds coherently, a fixed frequency offset from the receive carrier. The DSMCS will then record the received energy for retransmission to the ground processors along with the transmitter oscillator record. The ground processor recognizes rf lock in delayed time when the local phase-lock loop sweeps in synchronism with the transmitter oscillator including bias errors such as transmission time.

Once rf lock is verified the ground oscillator is programmed to slowly reduce the sweep rate and width and move toward a convenient operating frequency. This frequency is selected so that the doppler count at the processor remains within the range of the counters available. If a noncoherent demodulation is implemented for the AGC and antenna error signals at DSMCS, this frequency offset could be very critical. The program oscillator is tracked at the ground station back through the spacecraft until a stable point is reached. A range check is then made by initiating the basic range clock modulation. Some doubt exists as to whether or not any significant improvement in reliability or flexibility can be achieved by putting the range code generator at the ground station. Ranging would be accomplished at the ground at the expense of increased acquisition time. The spacecraft transponder would repeat the range modulation until the range loop locked onto the clock bit which would produce a range ambiguity once every clock bit. By successive additions of suitable components of the range code, the ambiguities would be eliminated. Following a complete range lock, the local range stream is periodically shifted to maintain correlation. The shifting rate is recorded periodically for velocity tracking. After acquisition is complete, range codes from DSMCS are limited by the 5 minute command period or a .6 AU range ambiguity. The last step in the acquisition procedure is correcting the antenna boresight. The error signals are filtered and evaluated along with the spacecraft trajectory and antenna servo loop dynamics to produce appropriate control signals for the servo system.

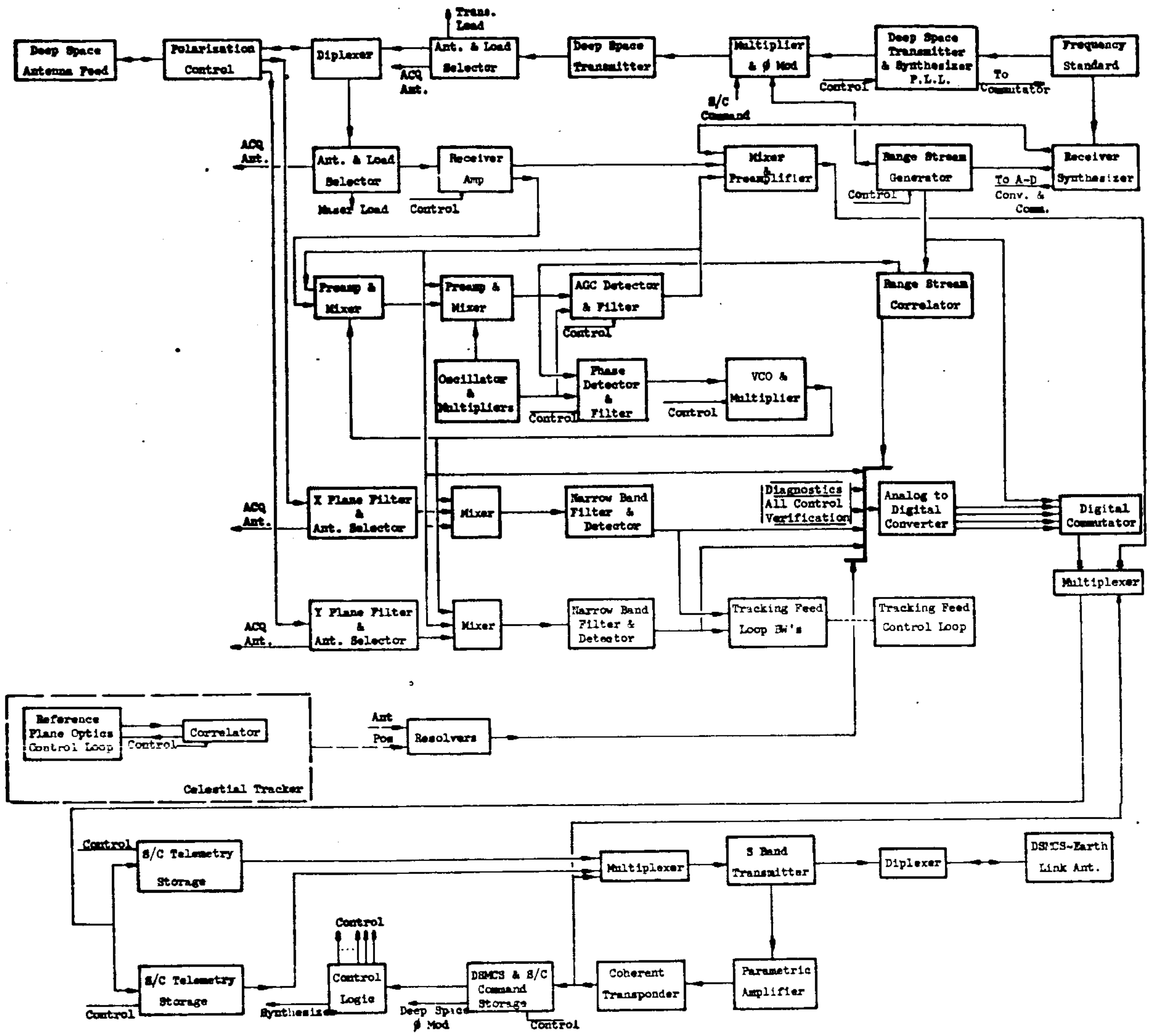

Figure 4.2-1. Communication System Block Diagram

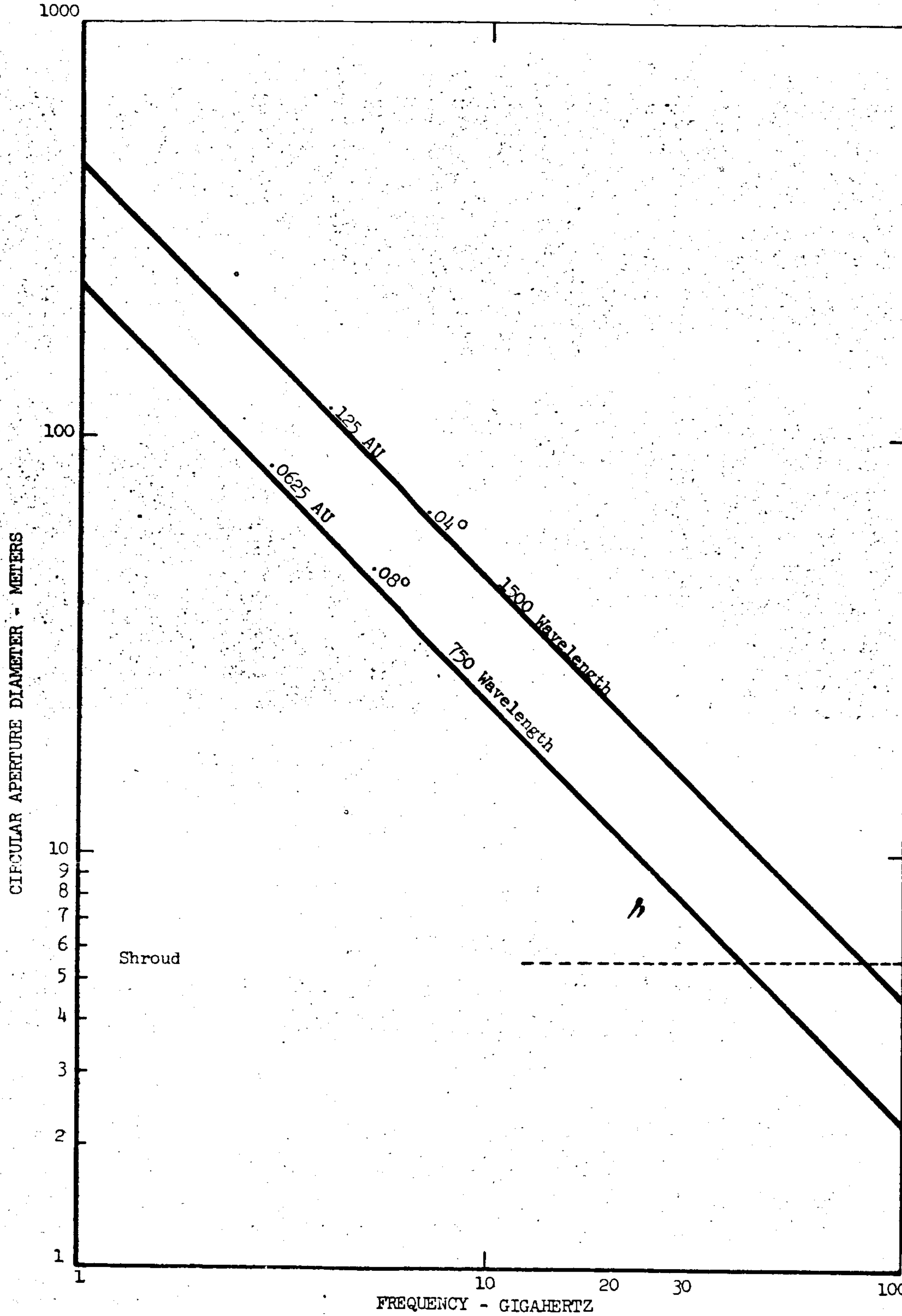

Figure 4.4-1. Aperture - Upper Limit

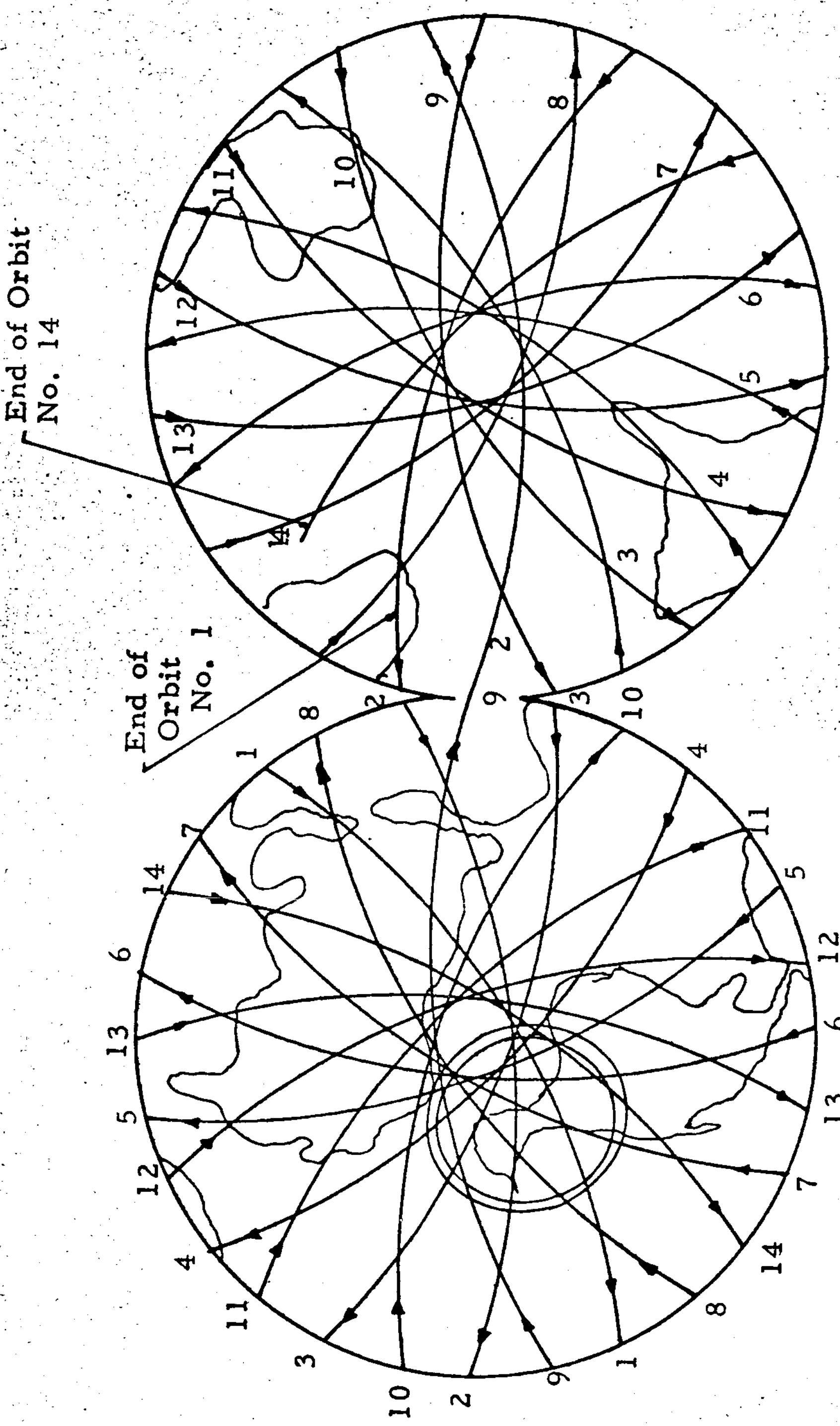

Figure 4.4-2. Satellite Orbits 80° Inclination

5.0 DISCIPLINE TRADEOFF ANALYSIS - PERFORMANCE VS WEIGHT/VOLUME WITH INTERFACE CONSIDERATIONS

5.1 INTRODUCTION

The primary goal of the tradeoff analysis is to establish a preferred satellite configuration. Secondary objectives are to point out the critical areas and to show how subsystem interrelationships affect the total system performance. The goodness criteria for maximizing the total system performance is the channel capacity from the deep space spacecraft to the DSMCS. The functional relationships which were taken from the many disciplines already discussed and which lead to a preferred satellite configuration are included in Figure 5.1-1. The configuration evolved from a method similar to Figure 5.1-1. The actual method required considerably more interrelationships which are not shown. If included they would only confuse the essential operations described in the figure. These further relationships should become clear as the results are presented.

The ranges of the parameters considered in the tradeoff were restricted either by limits established by the customer or matters of practicality. The frequency range was from 2 to 100 gigahertz. The payload mass and operating altitude were constrained by the capability of two boosters, Titan 3C, and Saturn 1B. The shroud diameter sizes 4.0 meters (13 feet) and 4.6 meters (15 feet) for the Titan and 6.1 meters (20 feet) for the Saturn represent the largest solid antenna sizes possible. The equipment state-of-the-art was extended to be compatible with the 1975 system design freeze although much of the equipment proposed is available now.

The spacecraft terminal of a deep space communication link was also considered in the computation of Figure 5.1-1. The most important assumption regarding the spacecraft was that its antenna is area limited. In addition, consideration was given to the spacecraft transmitter conversion efficiency.

The goodness criteria for this study was the channel capacity from the spacecraft to the DSMCS satellite. At least two other factors are just as important to achieve an effective satellite design, cost and reliability.

This effort was restricted to bit rate since it is the first study phase and is intended to investigate technical feasibility. Throughout the study, however, both cost and reliability were implicitly considered. For example, in the state-of-the-art equipment surveys the equipment specifications were generally extrapolated into the future from available commercial equipment without demanding a costly new device or technique in order to achieve the predicted specifications. Also, the subsystems selected for incorporation into DSMCS exhibited reliabilities in excess of one year at present or are anticipated to have reliabilities exceeding one year in the near future.

5.2 TRADEOFFS

Before reviewing the conclusions resulting from the iterative procedure described in Figure 5.1-1, some interesting subsystem relationships evolving from the procedure will be discussed.

5.2.1 ANTENNA

Antenna gain as a function of mass and frequency are shown in Figure 5.2.1-1 and 5.2.1-2 for the antenna types, single solid paraboloid and petal paraboloid. These configurations were selected because they provide maximum gain for the expected launch and orbital environment. The solid paraboloid is short of the thermal distortion gain limit for the Saturn 1B normal shroud size and the top study frequency 100 gigahertz. The petal structure is manufacturing limited before the shroud limit is reached for 100 gigahertz. By extrapolating the manufacturing and deployment art, a gain limit at 100 gigahertz will exist beyond the standard Saturn 1B shroud. However, the petal structure will be heavier than the solid for the same gain and diameter. It will be shown that the petal antenna is advisable for the Titan because of the shroud aspect. All the other antenna types studied with the exception of an array of solid paraboloids were found to provide much lower gain for the same operational environment. The array was not considered further because the improvement in gain over the solid dish is small and the feed problems associated with a common low noise receiver and transmitter are difficult at the higher frequencies.

The thickness of the single paraboloid and petal paraboloid antenna was found to be independent of the antenna size for the usable diameter range. Invar, which is required because of thermal distortion difficulties, average thickness will be 4.0 and 5.5 millimeters for the solid and petal antenna respectively. The effectiveness of the antenna, as a radiation shield can be seen in Figure 5.2.1-3 where the required shielding as a function of altitude for 1 and 2 years is shown. The antenna mass is adequate for all but proton effects at greater than 2 years and only above 1300 kilometers, for the solid antenna and 1650 kilometers for the petal antenna. The additional proton shielding not provided by the antenna may be accomplished by a local shield near the electronics where it is needed.

5.2.2 POWER SYSTEM

Initially, eight power profiles were recommended for the DSMCS. A 1.5 kilowatt and a 7.5 kilowatt continuous load were suggested to represent the general housekeeping power, and maximum cryogenic system plus housekeeping power, respectively. Peak powers of 30 kilowatts and 100 kilowatts were suggested for 5 minute/hour and 15 minute/day, for the purposes of command. Subsequent study has indicated that the higher command power energy requirements are prohibitive as is the 30 kilowatt command power for periods greater than 5 minutes due to thermal control difficulties. The 7.5 kilowatt maximum continuous load is primarily due to the cryogenic refrigeration system which may require up to 6 kilowatts power. This number is taken from existent systems of high reliability operating on the ground. Laboratory developments indicate that the refrigeration input power for space applications may be reduced to as low as 500 watts by 1975. Hence, a large uncertainity exists as to the required cryogenic refrigerator power consumption. On the basis of the preceding comments, profiles B and E (detailed in Volume III, Section 4.2) were selected to represent the most probable power profiles for the DSMCS. Profile B is a worst case model where an inefficient cryogenic closed loop refrigerator and a relatively high housekeeping load occurs. Profile E represents either an open loop cryogenic system and high housekeeping power requirements or an efficient closed loop refrigeration system and minimal housekeeping power requirements.

The energy requirements for profile B or E may be met by either an RTG or a solar cell power system. Stored chemical energy mass such as a fuel cell requires over 10,000 kilograms in one year for profile E. If the RTG and the solar cell power subsystems were compared strictly on the basis of the mass of each system the solar cell system would be superior by a ratio of 2:1. The minimum altitude at which the DSMCS may track the spacecraft is just below and in the lower radiation belts. The solar cell power system output as a function of time and altitude is shown in Figure 5.2.2-1 and profile B power level. In order to generate the power level demanded by the two profiles at the end of the DSMCS mission, surplus solar cells must be added. The extra power available with these cells would provide a safety margin for the earlier mission phases. The power supply battery is sized to utilize this power rather than dissipate it.

The comparison of the two types of power systems is difficult because of the complex interrelationships of solar cell deterioration, operation altitude, thermal radiator sizes and also the moment of inertia imbalance induced by the solar panels. The total mass of the two types of systems will be compared as a function of the mission duration. Figure 5.2.2-2 shows the basic solar cell power supply mass required in order that profile B power is available at the time indicated on the abscissa. The RTG mass is shown for comparison. The RTG is designed to dump excess heat generated by itself without additional thermal radiators. Thermal energy generated by the electronic package and cryogenic refrigerator output will be radiated in the RTG system or through radiating panels for the solar cell power system. The required radiating panel weight as a function of time for a 300° Kelvin radiator is added to the basic power supply weight in Figure 5.2.2-2. As the panels increase in size the ACS system mass increases rapidly. Details regarding the ACS mass will be reviewed in another section. However, the ACS mass which is directly accountable to the increased moment of inertia due to the larger panels has also been added in the figure. A comparison of RTG mass with the total solar cell system mass indicates clearly that the RTG mass is lower above 1,000 kilometers altitude for periods as short as one year. A similar plot of the basic power supply and component weights chargeable to the power supply for profile E is shown in Figure 5.2.2-3. Since the panels are shorter in profile E, the ACS does not become a problem as quickly as in profile B; however, the cutoff point remains about the same. Above 1100 kilometers, the RTG is lighter than the solar cell system for greater than one year lifetimes.

The weight of the antenna may be traded with that of the transmitter power supply in the command mode or with the cryogenic receiver weight in the receive mode. In either case the intent is to minimize the satellite mass for a given performance level. At the shroud limit the solid paraboloid antenna exhibits a gain slope of .25 db per 100 kilograms mass added to the satellite. Including transmitter conversion efficiency the improvement in power level for the same increase in mass of the transmitter is less than 1/3 that for the antenna. This is on the basis of a solar cell system at 750 kilometers altitude operating for one year, which is the most optimistic case. Thus, for maximum command capability the system weight is best invested in the antenna structure. Further, the primary DSMCS mission objective is to receive, which requires the maximum collecting aperture. Before a comparison may be made between the weight of a receiver and associated cryogenic equipment and the antenna, a comparison of the effectiveness of a closed loop with an open loop cooling system is required. The probable receiver is a cooled maser amplifier followed by a detector. Present maser amplifier noise temperatures and gain deteriorate very rapidly above 4.2° Kelvin. Parametric amplifiers are expected to operate continuously over a large temperature range down to liquid helium temperatures but unfortunately work above 20 gigahertz is very new and uncertainty exists as to whether they will ever operate effectively at 100 gigahertz. Thus, it must be assumed that a 1/2 watt heat load at 4.2° Kelvin in present maser systems is necessary for receiver operation.

A comparison of the various component masses necessary to supply the 1/2 watt heat load at 4.2° Kelvin is included in Figure 5.2.2-4. The load may be supplied by either a large Dewar or a closed cycle refrigeration system which, of course, must be backed up by an energy source. A Dewar plot includes the necessary helium for extracting heat by vaporization, Dewar insulation sufficient to reduce radiation loss to negligible levels and leak-off due to conduction losses. The solar cell and RTG plots represent the mass required to supply the energy to a six kilowatt closed cycle refrigerator. Clearly, the Dewar mass exceeds that of the RTG and the solar cell system with one exception beyond 1/2 year. Only when the solar cell system is in the high radiation region at 1650 kilometers does its mass become greater than that of the Dewar. More efficient closed cycle systems represented by the 3 kilowatt RTG plot would simply shorten the time during which the open loop system is superior.

5.2.3 ATTITUDE CONTROL SYSTEM

The attitude control system was one of the most sensitive satellite subsystems to the total DSMCS configuration. The stabilizing torques to counteract magnetic field force, aerodynamic pressure force and solar pressure force were found to be small compared to gravity gradient torque. The strong configuration dependence is due to the very large moment of inertia caused by the deployed solar panels. The ACS system for the 2 year, profile B RTG configuration mass is 435 kilograms (956 lbs) at 750 kilometers, decreasing above this point as the cube of the distance from the center of the earth and thus is quite reasonable. From what already has been discussed it should be clear that the area of the solar panel is a function of the power profile and the DSMCS mission duration and altitude. The smallest solar cell area is required for profile E, 750 kilometers altitude, and a one year mission duration. If the panel assembly could be implemented to produce a 10 meter wide panel, this particular case would require a panel 10 meters long also. This configuration establishes a strong moment of inertia in two planes. The moment in the plane orthogonal to these two is an order of magnitude lower. Because the gravity gradient torques are proportional to the difference in the moments of inertia along orthogonal axes, the ACS size to balance this relatively small solar panel is prohibitive for the very high DSMCS stability requirements. The ACS required to counteract the gravity gradient torque may be substantially reduced by balancing the moments of inertia along the three rectangular orthogonal axes. By projecting a fixed mass such as the battery in a direction normal to the plane formed by the axes of the larger moments of inertia, a low difference in moments of inertia may be achieved. The deployed mass must be fixed in position relative to the center of mass in order to maintain the balance. All of the solar cell configuration designs employed this technique to achieve balance and therefore reasonable ACS size. The length of the boom was adjusted to match the moments and the available battery mass.

The ACS mass for the balanced (equal moments of inertia along each rectangular orthogonal axes) satellite is given for profiles B and E in Figures 5.2.3-1 and 5.2.3-2. The panel width was set at a rather optimistic 10 meters. A double fold from either side of a 3.3 meter panel is a suggested method of deploying this array width. A narrower panel would raise the moment of inertia very rapidly since it is proportional to the length of the array squared. For example, the B profile ACS's mass would nearly quadruple for a five meter wide panel. The initial maneuver fuel is less than 5% of the first year's consumption on a 500 kilogram ACS. A redundant system is assumed for all the ACS masses given. Separate fuel containers and valve systems are incorporated to maximize the ACS reliability. Balanced ACS systems of more than 1,000 kilograms require boom lengths normal to the solar cell plane greater than 10 meters. Booms beyond this length are possible, however, the deployment and thermal bending once deployed become serious problems. The boom weight relative to the battery also becomes appreciable in order to maintain the battery location relative to the center of mass beyond 10 meters. Balance problems also occur for the solar panels because when the panels are rotated, balance is upset. One solution is to rotate the boom and weight to maintain its relation normal to the solar cell array but this is preferably avoided if at all possible. Another approach is to permit the tracking antenna to move with respect to the satellite. This requires sensors and ACS on the antenna and to a lesser degree on the satellite.

The position of the tracking antenna and the ACS system nozzles relative to the star tracker will vary due to the thermal and mechanical loads on the structure. With careful design, a maximum alignment error of $.05^{\circ}$ is anticipated. This is comparable to the smallest beam width and thus would contribute to trajectory calculation error and search time during initial acquisition. With flight experience this error may be taken out by establishing tracking antenna bias as a function of satellite altitude and orientation with respect to the sun. The bias may be evaluated periodically by tracking known radio sources near the direction of the spacecraft.

5.2.4 SPACECRAFT AFFECTS

Although deep space spacecraft have not been configured in detail for post-1975 missions, some generalizations can be made regarding their characteristics as a deep space communication terminal. An area limited spacecraft antenna is assumed here. The shroud size of the probable boosters needed for the spacecraft preclude gain limit antennas as long as the proper materials and antenna types are implemented. Solid surface antennas such as suggested by DSMCS would be used. The frequency dependence of an aperture limited antenna at both terminals using the expected conversion efficiencies of a 100 watt plus transmitter is shown in Figure 5.2.4-1. The expected transmitter efficiency for the 1975 era is seen to be high. A spacecraft ACS will, of course, be a serious problem if the antenna size and operating frequency push the gain to 60 decibels. The spacecraft ACS mass is not comparable to that of the DSMCS since the dominant force fields near the earth are negligible in interplanetary media. A detailed analysis of the spacecraft ACS problem is beyond the scope of this study. However, effort will be made to keep the operating frequency as low as possible if it does not compromise channel capacity in order that the spacecraft ACS mass is minimized.

5.3 RESULTS

The preceding discussion describes the tradeoffs whereby specific subsystem hardware and characteristics were selected in order to achieve compatibility with each other and to minimize overall mass and to maximize receive channel capacity. The major DSMCS system parameters, including the operating frequency, satellite altitude, satellite weight and booster type, are selected here as a function of the mission of the spacecraft to be tracked.

5.3.1 FREQUENCY

The study frequency range is 2 to 100 gigahertz. The components which affect the operating frequency selection are the spacecraft transmitter and antenna, the DSMCS antenna and receiver, and the frequency stability of the source oscillator and that of the detection oscillator. In the case of both the satellite and the spacecraft antenna, the reflector surface is assumed to be sufficiently smooth that the antenna is not gain limited over the frequency range of interest. The relatively small size of the spacecraft antenna indicates that the antenna would be gain limited beyond 100 gigahertz if a solid dish is used. The solid satellite antenna is just short of the gain limit for the standard Saturn shroud diameter (6.1 meters). This is also true for the satellite petal antenna if manufacturing technology is extrapolated into the 1970's. An area limited antenna at both terminals produces a communication link channel capacity which is proportional to the frequency squared. Thus, an apparent advantage of 34 decibels occurs between a 2 gigahertz and a 100 gigahertz system. Other factors reduce this advantage due to the general decrease in effectiveness of equipment as the operating wavelength becomes small compared to the components.

A survey of the candidate spacecraft transmitters in this frequency range available for the 1975 era indicate that the frequency dependence is such that the transmitter conversion efficiency is four decibels poorer at 100 gigahertz than at 2 gigahertz. The satellite system noise temperature is expected to range from 25°K to 40°K for the 2 to 100 gigahertz range. This reflects in a system noise temperature increase of two decibels. The channel capacity loss induced by receiver and transmitter frequency sensitivity over the range is six decibels. When compared with the 34 decibel advantage which occurs from the higher antenna gain at 100 gigahertz, it is clear that a 100 gigahertz system provides the maximum channel capacity. There are, however, two other equipments which must be considered, oscillator stability and both the satellite and spacecraft attitude control system. The satellite ACS for the 100 gigahertz system is of reasonable size except for the extreme case of very large solar panels. The ultra stable platform with a 100 gigahertz system provides the low angular tracking error needed for future deep space missions. Little can be said about the spacecraft ACS, since the size of the system is strongly related to the yet-to-be-designed spacecraft. The ACS system will definitely require an improvement in the present state-of-the-art of spacecraft attitude control system.

Local oscillator phase noise of a phase lock loop system for a given oscillator stability is directly proportional to the operating frequency and inversely proportional to the phase lock loop bandwidth. The phase lock loop bandwidth is usually made sufficiently narrow in order to keep the receive carrier above the threshold and broad enough to prevent the phase noise of the local oscillator from becoming appreciable compared to the input noise. Present oscillators which are controlled by atomic standards for long term stability produce short term phase noise of .075 radians rms in a ten hertz rf bandwidth at 10 gigahertz. By extrapolating oscillator stability trends from 1958 to 1966, a similar phase noise should be exhibited in 1975 for a 100 gigahertz system in a ten-hertz bandwidth. The long-term stability offered by the atomic standard itself is excellent. The phase noise which is usually kept an order of magnitude below that due to thermal noise is induced by the source transmitter oscillator as well as the receiver oscillator. Since it is not practical to close the DSMCS spacecraft loop in real time from DSMCS, the burden of achieving both long-term and short-term stability falls on the spacecraft oscillator. DSMCS will, of course, incorporate such an oscillator. Thus, a 100 gigahertz carrier may be implemented in a deep space channel, but it does require a relatively sophisticated oscillator at the spacecraft.

Recent research in detection theory indicates that a substantially lower spacecraft transmitter power will be necessary for a one-way communication channel compared to a two-way closed loop channel due to the reduction in detection efficiency at the earth terminal receiver caused by the noisy carrier at the spacecraft in the two-way system. It is shown further that when the modulation index (portion of total power in the carrier) is selected for minimum bit error rate, the noise on the carrier derived from the earth to spacecraft link causes an irreducible error on the spacecraft to earth link. This may be remedied by employing a clean stable oscillator at the spacecraft or raising the command power. The necessary increase in command power is a function of many loop parameters. For a typical two-way system where the spacecraft to earth channel bit error rate is 10^{-3} the carrier to loop noise at the spacecraft must be greater than 16 decibels.

In summary, the preferred operating frequency is 100 gigahertz. The preferred frequency is relatively independent of the spacecraft mission. Design specifications, particularly in the spacecraft area, must be upgraded to accommodate the higher frequency; however, such an upgrading is considered within the extrapolated state-of-the-art.

5.3.2 ALTITUDE

The total satellite mass available as a function of the altitude is shown in Figure 5.3.2-1 and 5.3.2-2 for a Saturn IB booster. Both eastern test range and western test range launch sites are shown, although there is at present no Saturn IB launch capability at the western test range. Such a capability could, of course, be achieved if cost justified it. The abscissa marks indicate the minimum altitude at which a DSMCS orbit may be inclined to produce continuous line-of-sight tracking to the spacecraft mission indicated. The payload is compared to the booster capability in the two figures for the profile B and E, since these profiles represent the extremes of the range of electrical power required.

The Mercury spacecraft mission examined entails an altitude of 842 kilometers for continuous line-of-sight tracking and 115-day mission duration. The relatively short mission duration suggests the use of a Dewar for the cryogenic system if six kilowatts are required to provide the equivalent cooling. A more efficient closed loop system using only three kilowatts would weigh less. From Figures 5.3.2-1 and 5.3.2-2 the minimum mass satellite utilizes a solar cell power system for both profiles considered. The allowable satellite mass and altitude combination for DSMCS tracking the Mercury spacecraft is bounded on the left by a vertical line at the abscissa altitude mark, on the top by the booster capability lines for the respective test ranges and below by the satellite mass plots. The larger operating region for profile E reflects the lower power subsystem weight and its consequences as reflected on other subsystem weights. If the operating DSMCS altitude for the Mercury mission remains at the minimum altitude, the additional payload may be absorbed by science instrumentation which could effectively exploit the ultrastable platform or the additional mass may be invested in more reliable equipments. Additional margins such as attitude control system fuel or solar panel area could quickly absorb the extra payload capability. If instead of operating at the minimum altitude the satellite is raised into the lower radiation belt, the RTG configuration weighs less than the solar panel. A completely redundant DSMCS is possible at the minimum altitude with both profiles. Further examination of launch windows and necessary longitude of ascending mode may indicate that a pair of DSMCS may be launched with a single Saturn booster in such a fashion that two different spacecraft missions with different regression rates may be tracked. Additional payload surplus may be invested in an orbit maneuver system. The noise temperature of Mercury is quite low so that at encounter the system noise contribution by the planet is small. However, the sun is expected to degrade performance because of its high noise temperature midway through the tracking mission. The relatively higher gain afforded by the DSMCS antenna will reduce the effect of the sun on the communication link.

The Mars mission duration of 1.23 years and minimum operating altitude of 1,116 kilometers suggests either that an RTG or a solar cell powered satellite may be used with the Saturn booster. The planet system noise temperature contribution is below one degree Kelvin at encounter and therefore is of no consequence to the total system noise temperature. Occultation of the spacecraft by the sun will occur near the end of the return voyage. The effects of the sun will again be reduced because of the relatively high gain of the DSMCS antenna. The satellite altitude and payload margin area in this case is smaller than the Mercury mission because the minimum altitude is greater and the mission duration is significantly larger. A redundant DSMCS may still be implemented for the E profile particularly with a western test range launch.

Three Jupiter missions were reviewed. A slow transit mission in 1978 and two fast transits in 1973 and 1978. The noise temperature contribution of Jupiter at encounter in all three cases was approximately $12^{\circ}C$ because the range was near six A.U. in each case. The two fast missions, 1.86 years, indicate that the RTG system is preferred for the B profile and either power systems may be used for the E profile with the Saturn booster. The slow mission length of 2.9 years precludes solar cell power systems for either profile. A payload margin of 20 percent remains for the eastern test range even for the B profile. The long mission length presents very serious reliability problems for all the subsystems utilized.

The Titan booster capabilities and payload similar to Figure 5.3.2-1 and 5.3.2-2 are shown in Figures 5.3.2-3 and 5.3.2-4. A 6.1 meter (20 feet) diameter petal antenna is configured in the B profile plot and a 9.1 meter (30 feet) diameter petal antenna in the E profile. The petal antenna although less efficient in gain available per antenna mass was selected in order to provide antenna gain comparable to the Saturn configuration , within the narrow Titan shroud. The petal structure would unfold after achieving orbit. The largest circular Titan solid antenna diameter is 75 percent of that possible in the Saturn shroud and thus provides at least 2.5 decibels less gain.

The profile B Titan configuration was restricted to a 6.1 meter (20 foot) diameter petal in order to keep the satellite weight within the available payload for ETR. Profile B solar panels were not considered practical with a large petal antenna primarily on the basis of packaging difficulties. From Figure 5.3.2-4 it is clear that the profile B configuration can be accommodated for the Mercury and 1978 fast Jupiter missions by a Titan launched from ETR and for all missions launched from WTR. The maximum channel capacity for all 6.1 meter (20 foot) diameter petal antenna configurations will be 1.5 decibels below the single solid paraboloid on the Saturn.

The channel capacity of the E profile system incorporating the 9.1 meter (30 foot) petal paraboloid will be the same as the Saturn shroud limited paraboloid. The lowest operating frequency at gain limit would be near 60 gigahertz. The lower frequency is preferred in order to minimize spacecraft requirements such as ACS pointing accuracy. From Figure 5.3.2-4 it is clear that all missions may be carried by the Titan from ETR except the 2.9 year Jupiter mission and that a WTR launch could fulfill this mission.

5.3.3 CHANNEL CAPACITY

The channel capacity of any deep space communication link is a function of a number of qualifiers, some of which are peculiar to the spacecraft mission and some of which are purely judgment. In order to make an effective comparison between a space communication link using DSMCS and the present earth-bound terminals, a common reference spacecraft must be selected. It is preferred that this spacecraft exhibit the general characteristics of future spacecraft design, such as three axis stabilization, and also that sufficient flight experience has been acquired with it so that the design approach is verified. The Mariner IV spacecraft meets these criteria and was used as the reference. The signal design, of course, varied from one mission to the next, but the error probabilities and the threshold levels were comparable and thus the required power levels were also comparable.

The relative performance of a communication link from a spacecraft in deep space to either an earth-bound or an earth-orbiting terminal is shown in Figure 5.3.3-1. Included in the figure is the 85-foot-diameter and 210-foot-diameter deep space network station operating at 2.3 gigahertz. The recently-built 210-foot dish is gain limited at 3.7 gigahertz. Beyond 3.7 gigahertz, the gain of the present 210 foot antenna decreases as the inverse fourth power of frequency. By improvements in the antenna reflector surface technology, a gain limit antenna near 10 gigahertz is anticipated. The reflector system performance at that frequency is also shown. The atmosphere model used in order to calculate the system noise temperature was 1.25 cm per hour rain and a ten degree elevation angle. This model is necessary to avoid the weather blocking out transmissions. Above ten gigahertz, the system noise temperature is increased by the ground radiation energy scattered by the rain. Also, the insertion loss becomes appreciable above four gigahertz as shown in Figure 5.3.3-2. System performance therefore deteriorates rapidly above ten gigahertz as evidenced by the 15 gigahertz point in the figure for the 210-foot-diameter, ten gigahertz gain limited antenna.

DSMCS performance for the solid antenna Saturn booster configuration is shown in Figure 5.3.3-1 at the frequency where it is greatest, 100 gigahertz. The DSMCS point includes a four decibel penalty for transmitter conversion efficiency loss relative to that at lower frequencies. The additional ordinate scales show the bit rate for a link between the terminals near the earth and a Mariner IV spacecraft at encounter with the terminals indicated. The Mars ordinate scale represents the Mariner IV 1965 capability at encounter. The planned 30-day-stay Mars mission bit rate is only .8 decibels higher. The Mercury bit capability at encounter is scaled next. The last ordinate scale describes the 1978 slow Jupiter mission capacity at encounter. The 1978 fast mission is .12 db bits per second lower and the 1973 fast mission is .44 decibels higher. All these rates could, of course, be increased by spacecraft improvements such as increasing the transmitter power above ten watts. The Titan boosted DSMCS point is identical to the Saturn boosted DSMCS for the 9.1 meter (30 foot) petal antenna and 1.5 decibels lower for the 6.1 meter (20 foot) petal antenna.

Study constraints restricted the frequency range from 2 to 100 gigahertz. Table 5.3.3-1 presents a comparison of present communications and DSMCS communication capabilities from spacecraft. The intent of the study was not to dwell on spacecraft improvements but the table presents obvious communication channel capacity improvement using 100 gigahertz hardware on spacecraft. The table illustrates more than a factor of 10 improvement in channel capacity by increasing frequency from S-band to 100 GHz for similar spacecraft equipment. Or for a 7.9×10^6 data rate, the S-band spacecraft example would require a transmitter output increase from 100 watts to 1000 watts.

5.3.4 OPTICAL COMMUNICATIONS

Opinions comparing microwave and optical space communications vary widely. Microwave technologies are on firm ground having proven hardware. Optical communication using lasers represents an exciting research field, however, it remains in the laboratory. Laser researchers working in the field of communications have varying opinions as to the predictable capabilities of laser. One opinion (Reference 1) states that a 1 meter aperture in space and a 10 meter aperture on the ground could provide a bandwidth of over 1 MHz with a 30 db signal-to-noise ratio at 1 A.U. This system was proposed to weigh between 450 kg and 675 kg. The power input requirement was suggested as a modest 200 watts. This system would provide a bandwidth of 50 MHz at 30 db SNR using a typical modulation scheme at a range of 0.4×10^6 km.

Another reference states that lasers open a new regime of spectral purity in the optical frequencies and a new regime in energy density and that lasers "must surely have a good future in scientific research." (Reference 2) The statement continues and notes radio-frequency systems are better than optical systems for earth-spacecraft-earth-communications.

Optical communication will not compete successfully with microwave systems for space applications until serious problems on the spacecraft are solved. Lasers are extremely narrow beam devices. Ultimate beamwidths of one second of arc or less are typical values, even when considering advanced techniques such as narrow band, high gain, noise free amplifiers and frequency modulation. The narrow beam is sensitive to optical aberation, propagation time, and refraction effect which compound the spacecraft beam pointing problem and the attitude control requirements.

A fair evaluation of microwave and optical communications would require investigation of the sensitivity of spacecraft disciplines to optical communication system requirements. A comparison on the basis of bits/sec/kg would be appropriate. Uncertainty in laser technology prohibits this type of comparison. From a consensus of opinions, it is agreed that microwave techniques will perform deep space communications in the immediate and perhaps distant future. There is however room for improvement in the microwave region as the intent of this study has shown.

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2. Radio and Optical Space Communications, P.D. Potter, R. Stevens, W.H. Wells, Jet Propulsion Laboratory, TM No. 33-85, October 30, 1962.

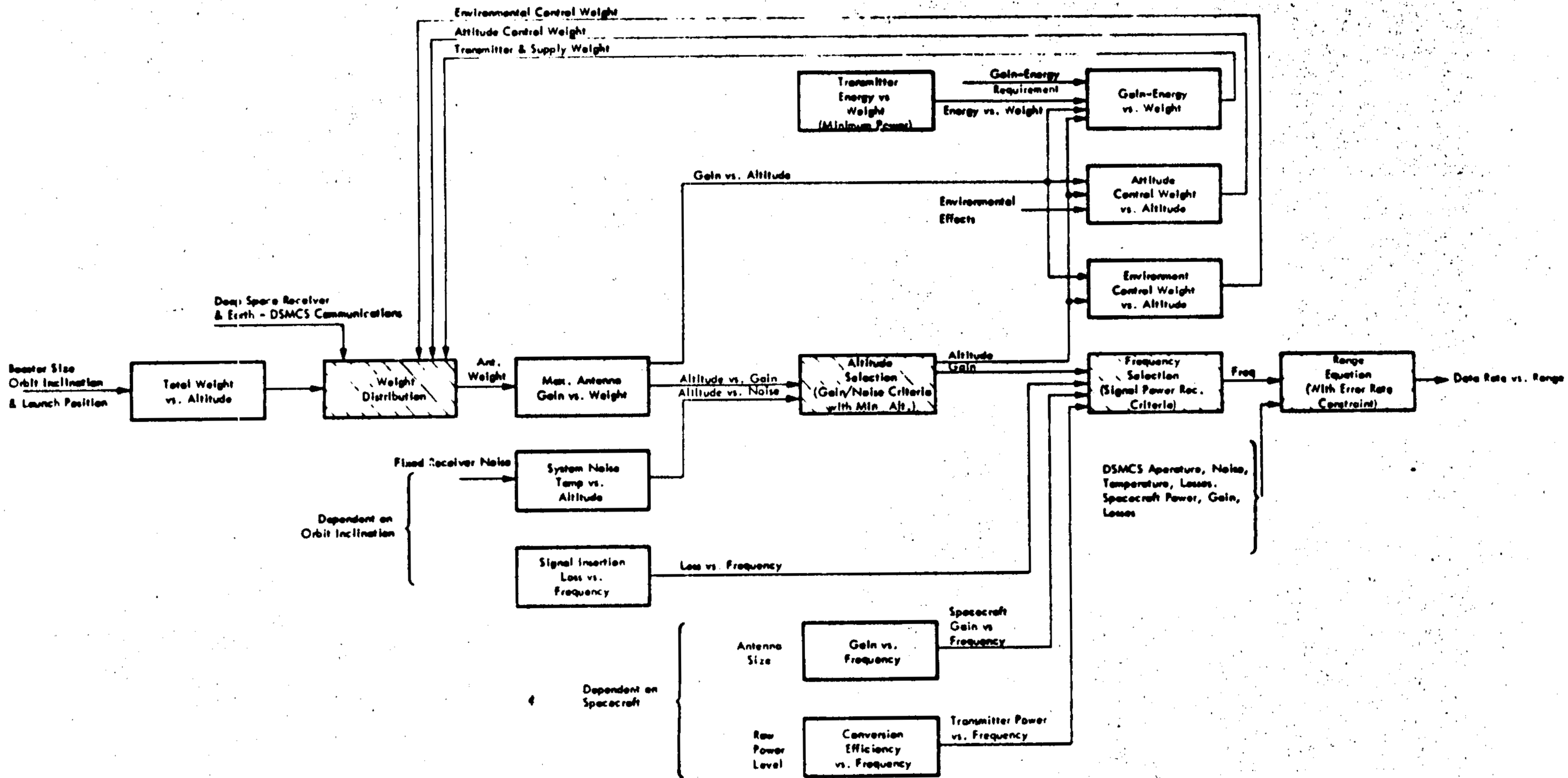


Figure 5.1-1. Communications Performance Analysis Plan

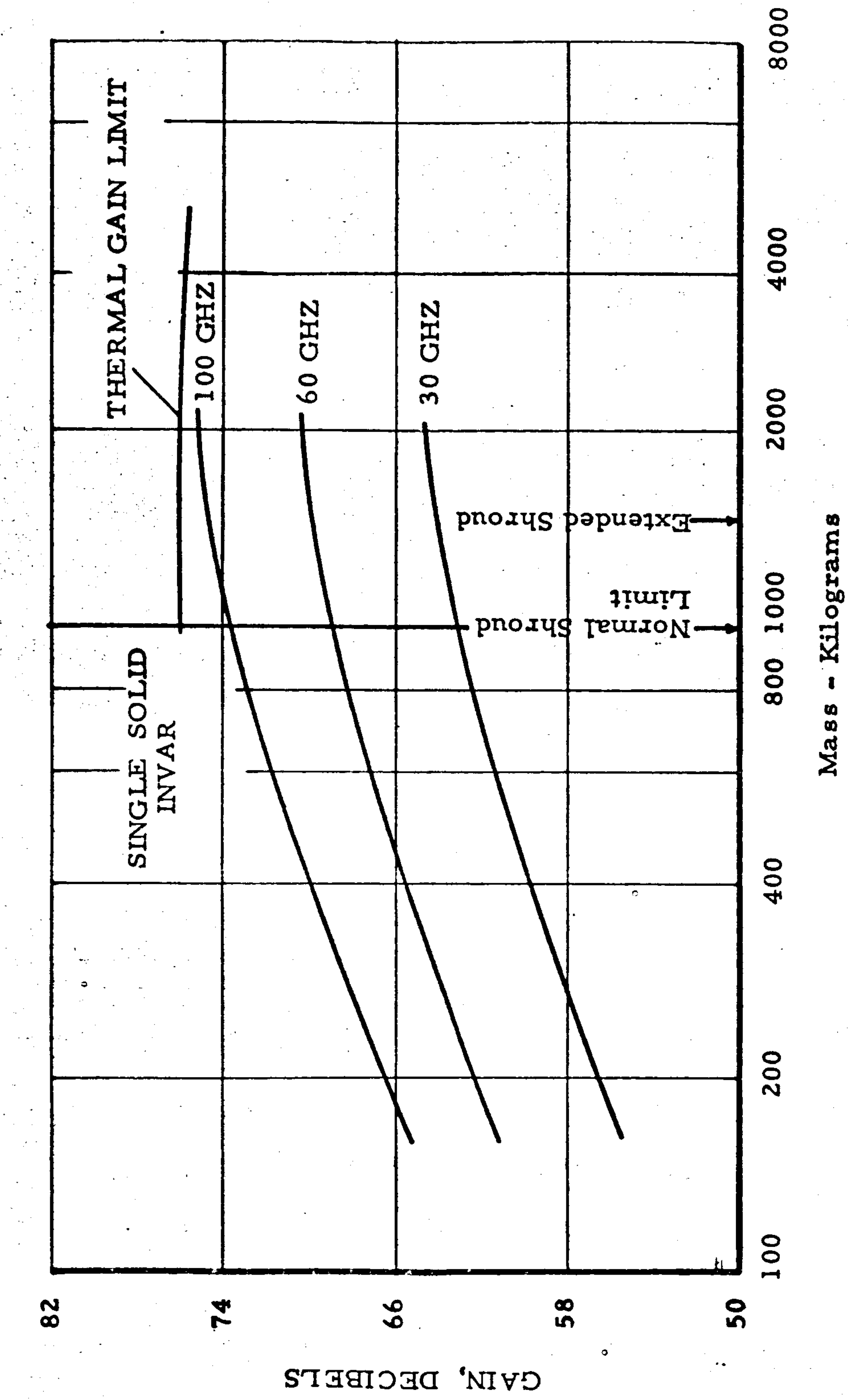


Figure 5.2.1-1. Solid Antenna Gain vs Mass

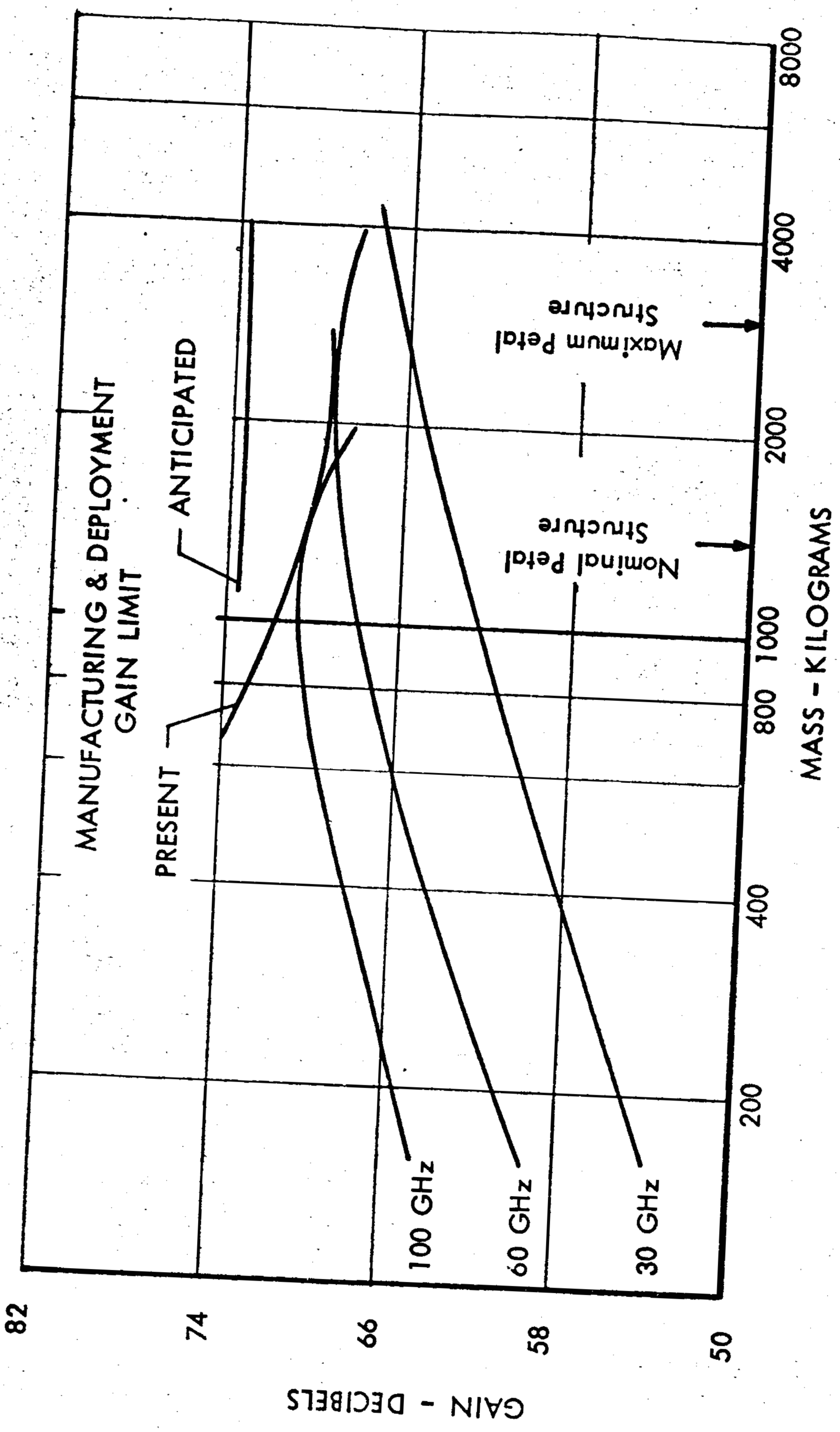


Figure 5.2.1-2. Petal Antenna Gain vs Mass

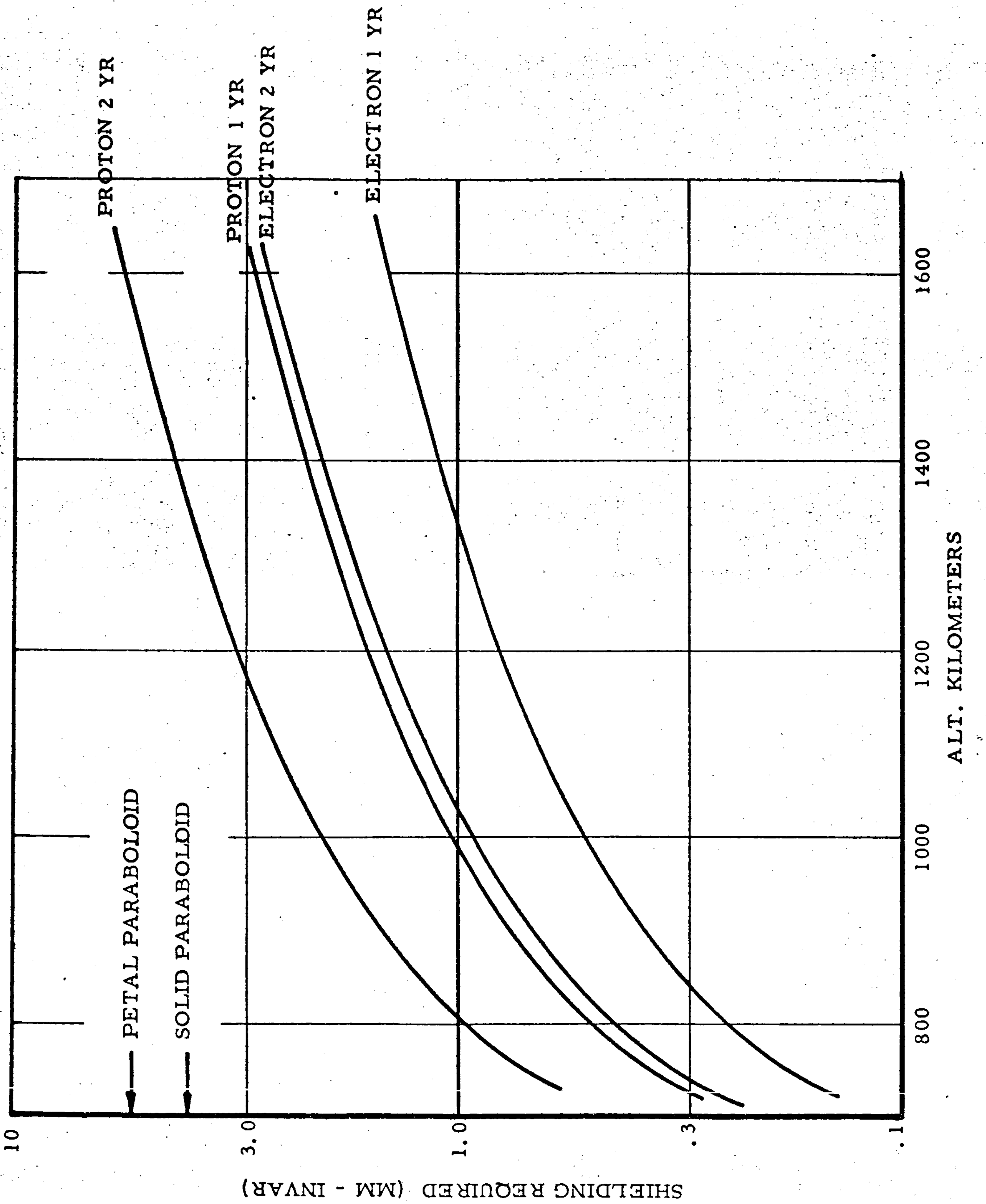


Figure 5.2.1-3. Shielding Requirements vs Altitude

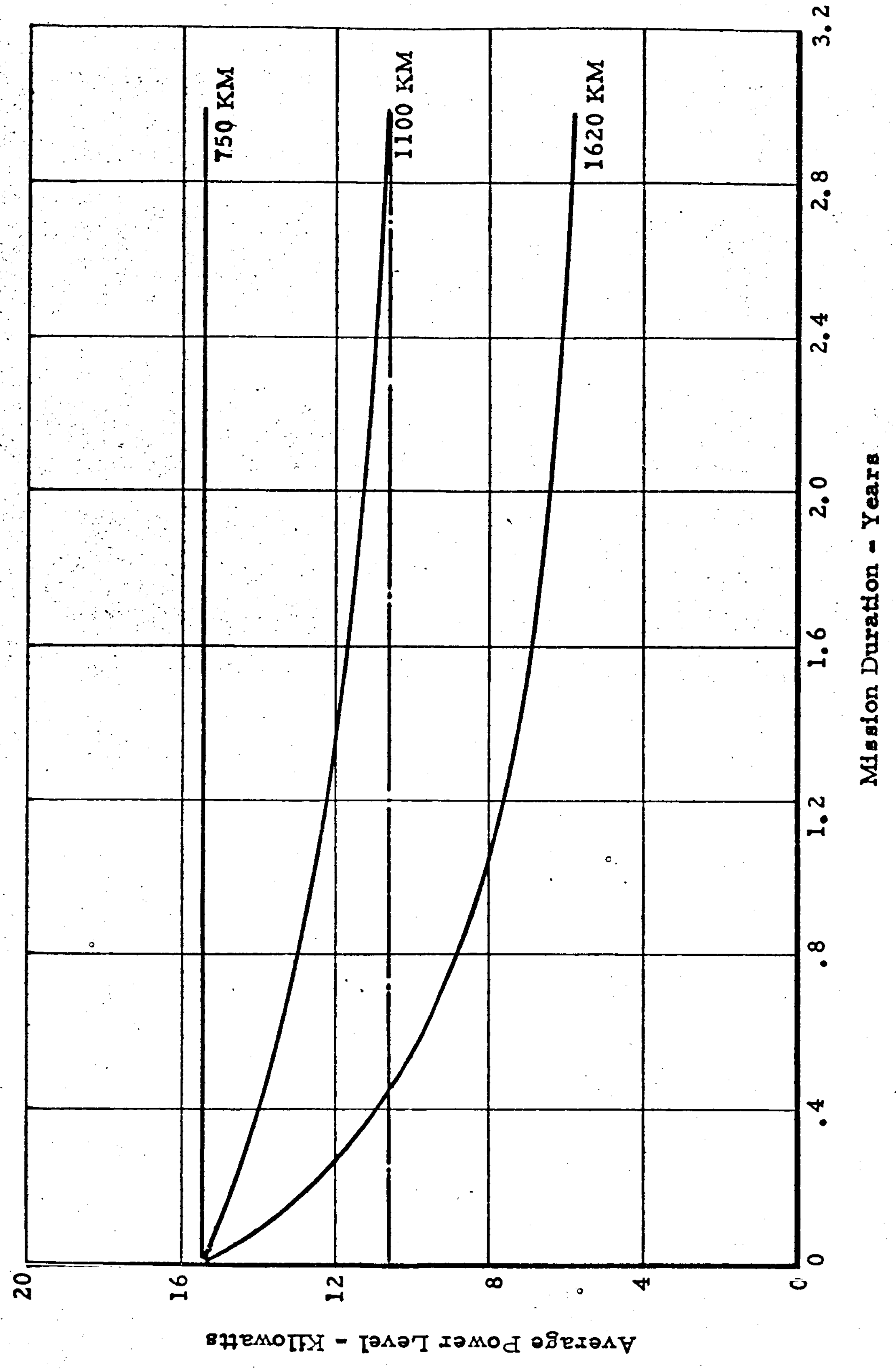


Figure 5.2.2-1. Solar Cell Power Relationships for Power Profile

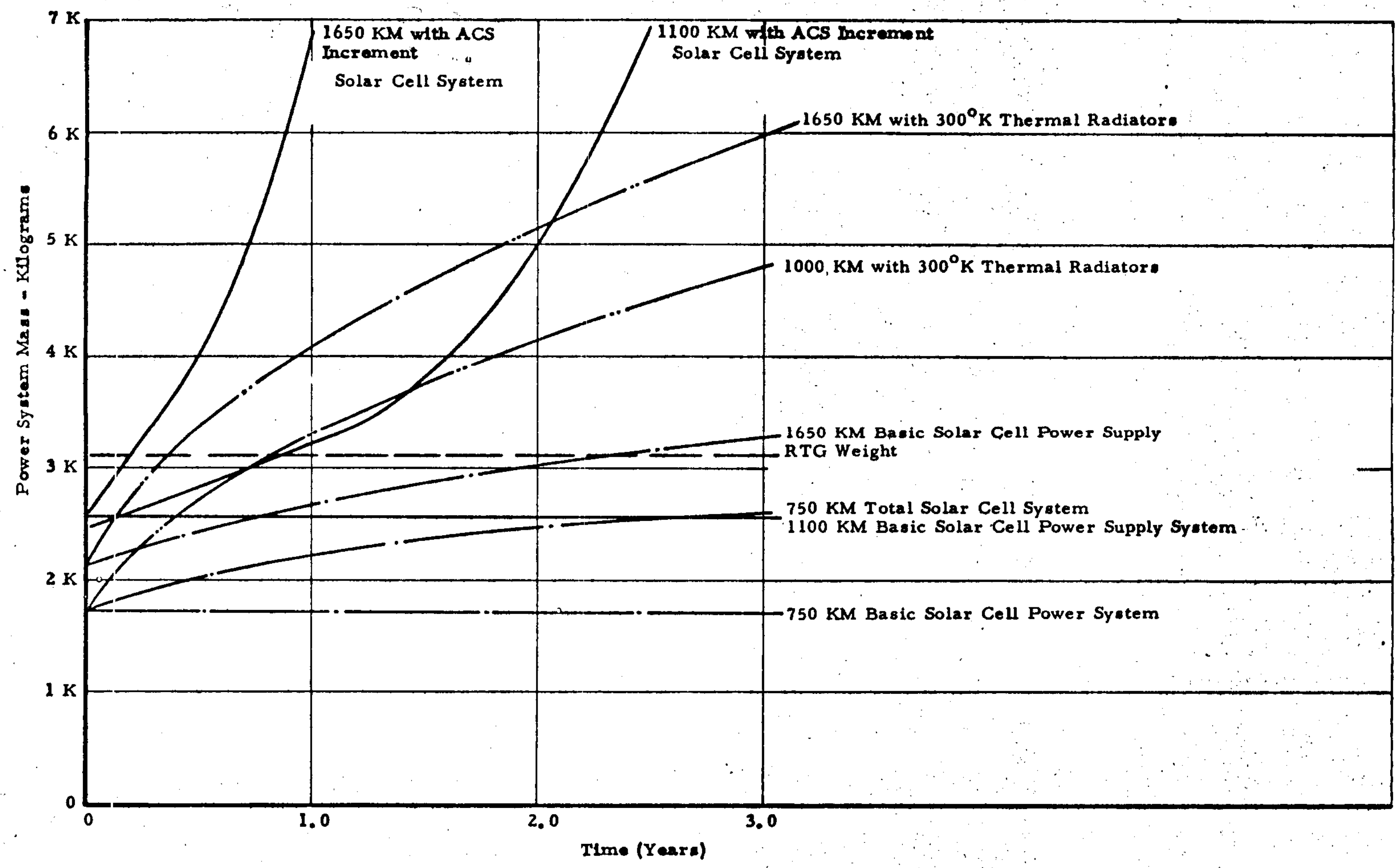


Figure 5.2.2-2. Power System Mass/Time/Altitude Relation

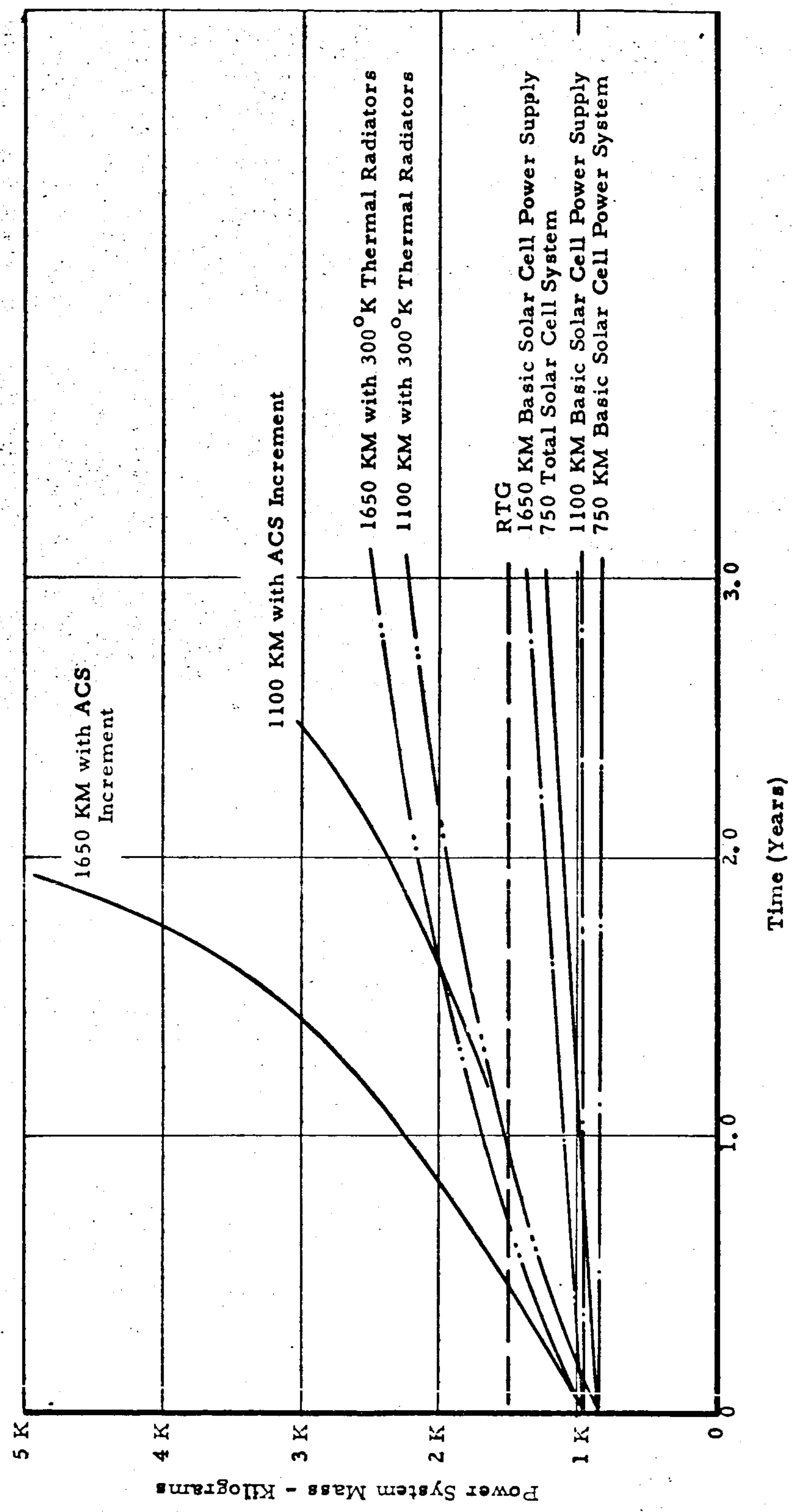
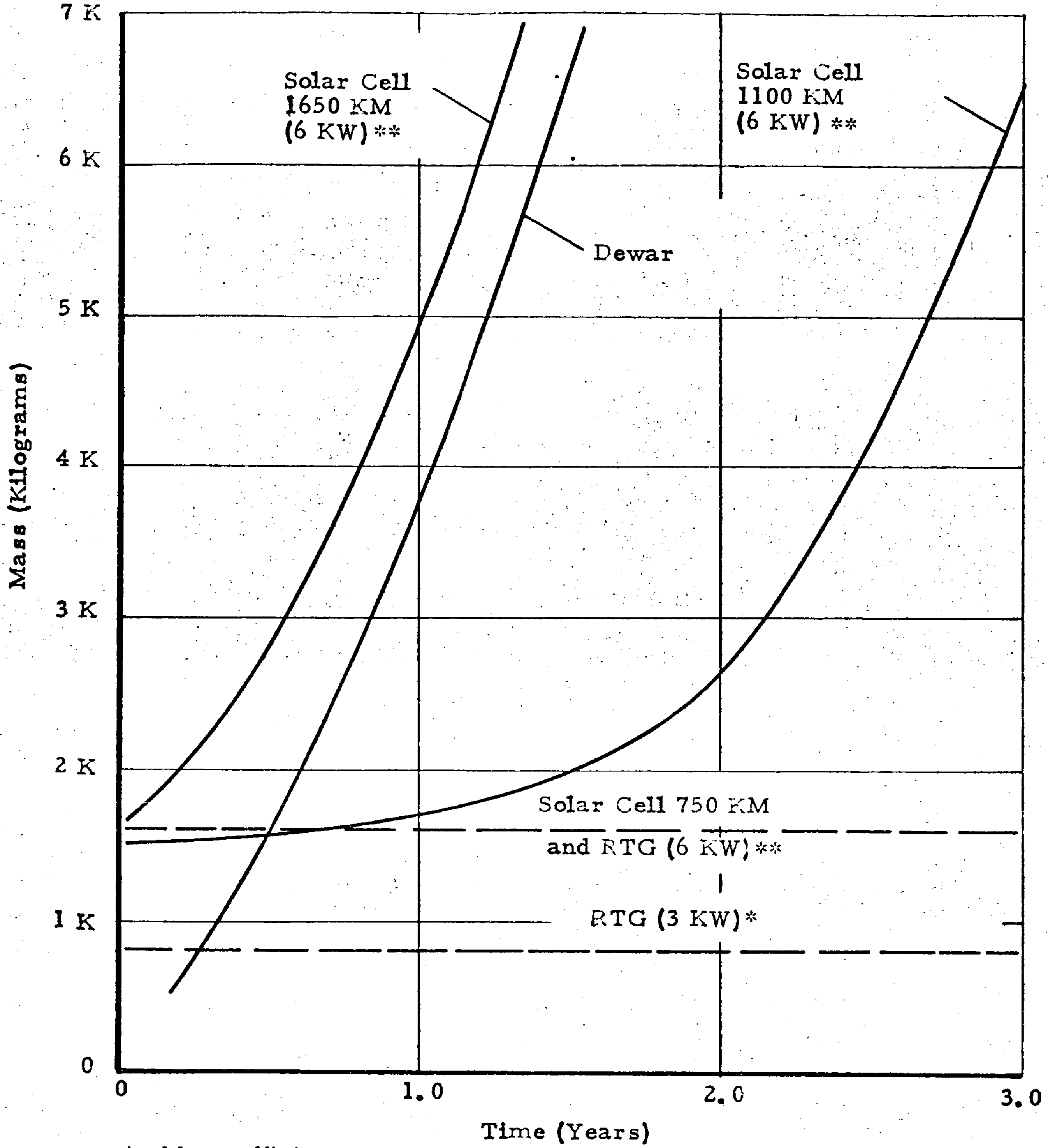


Figure 5.2.2-3. Power System Mass/Time/Altitude



* More efficient system
** Nominal efficiency

Figure 5.2.2-4. System Weight Required to Supply 1/2 Watt of Cryogenic Cooling at 4.2°K

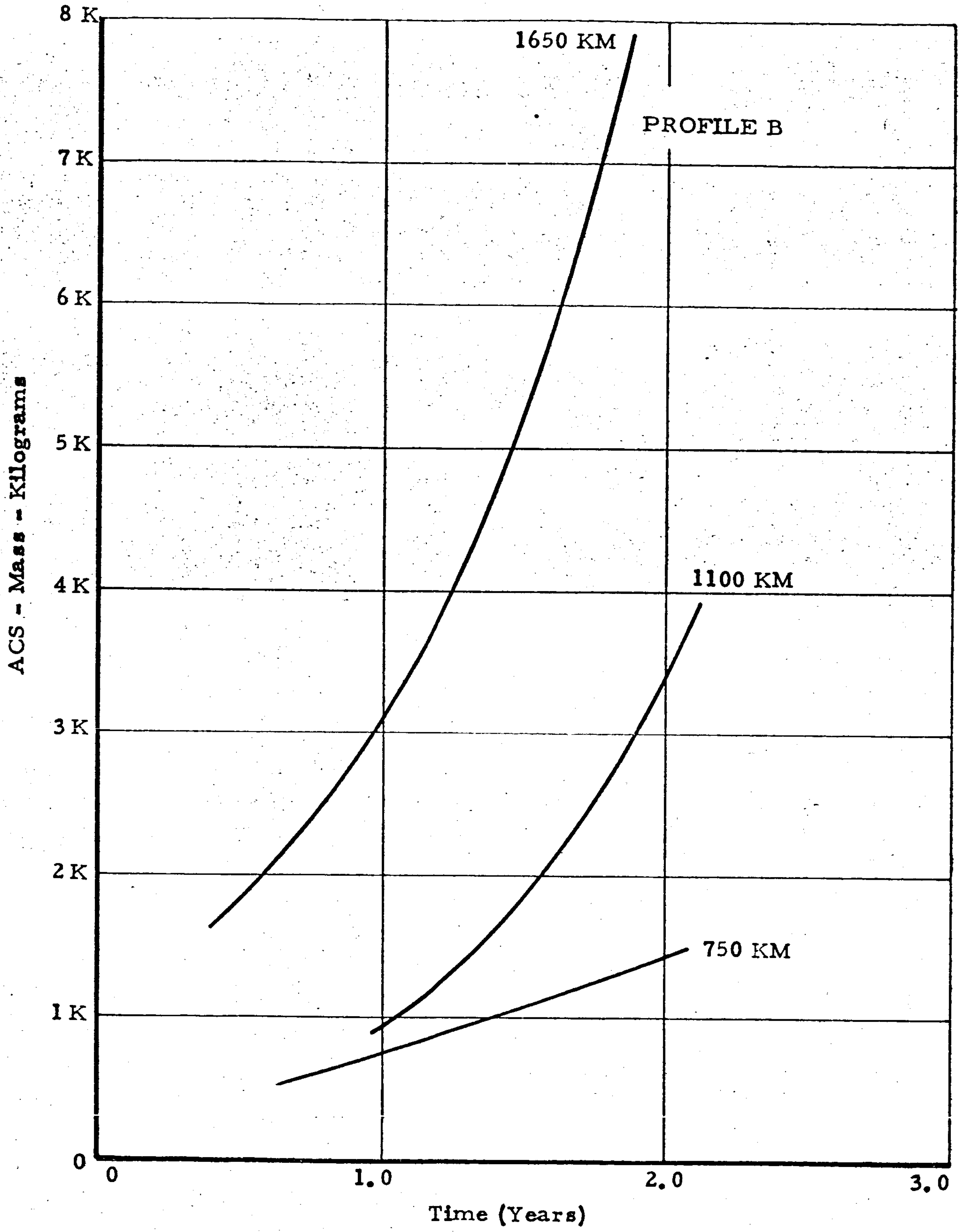


Figure 5.2.3-1. ACS Weight vs Time in Orbit-Profile B

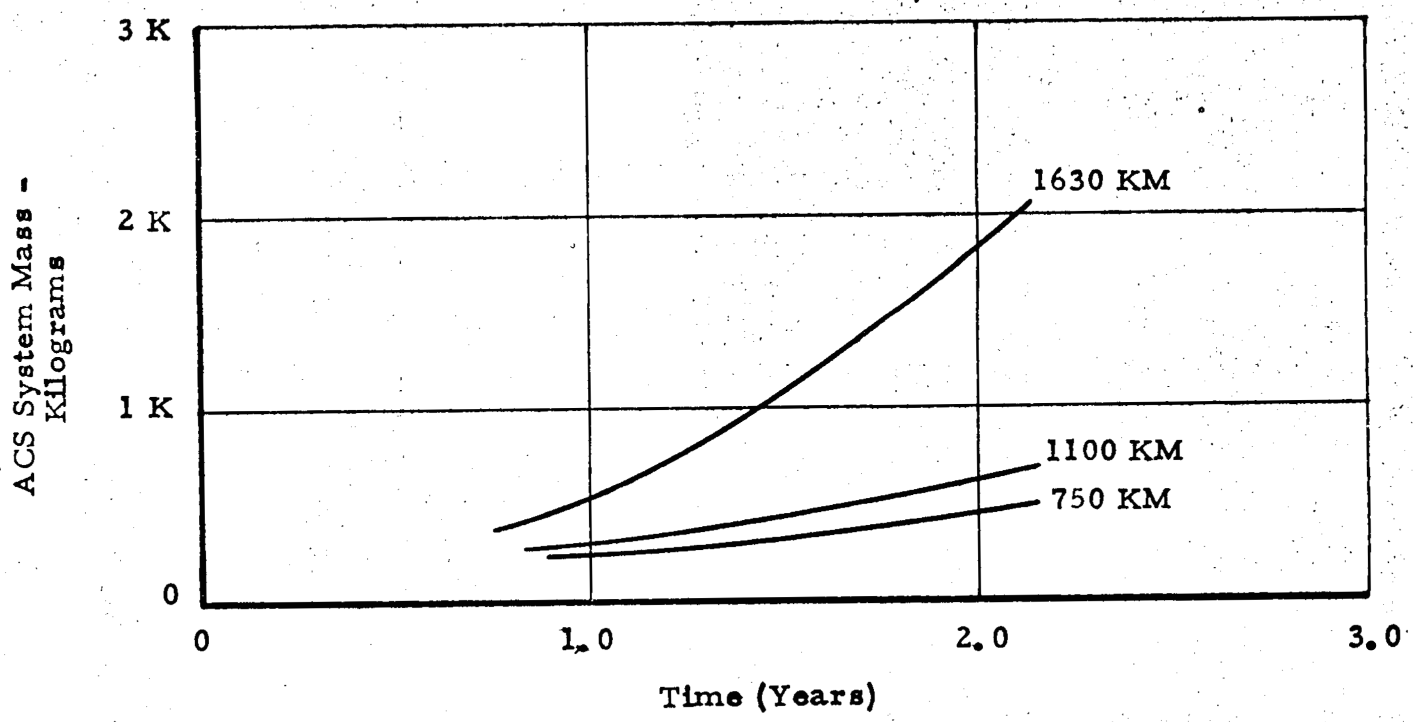


Figure 5.2.3-2. ACS Weight vs Time in Orbit Profile E

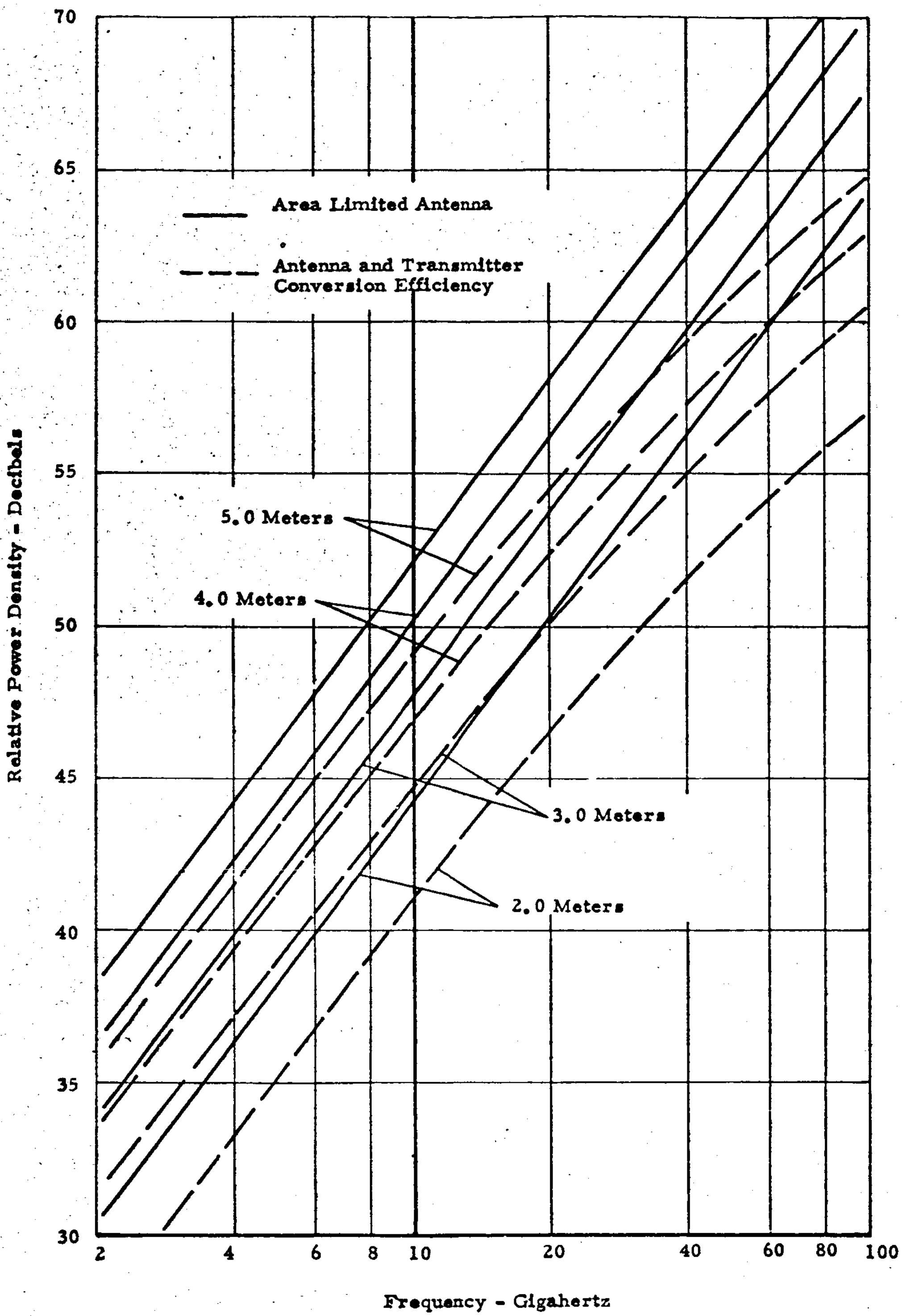


Figure 5.2.4-1. Relative Power Density vs Frequency for Spacecraft

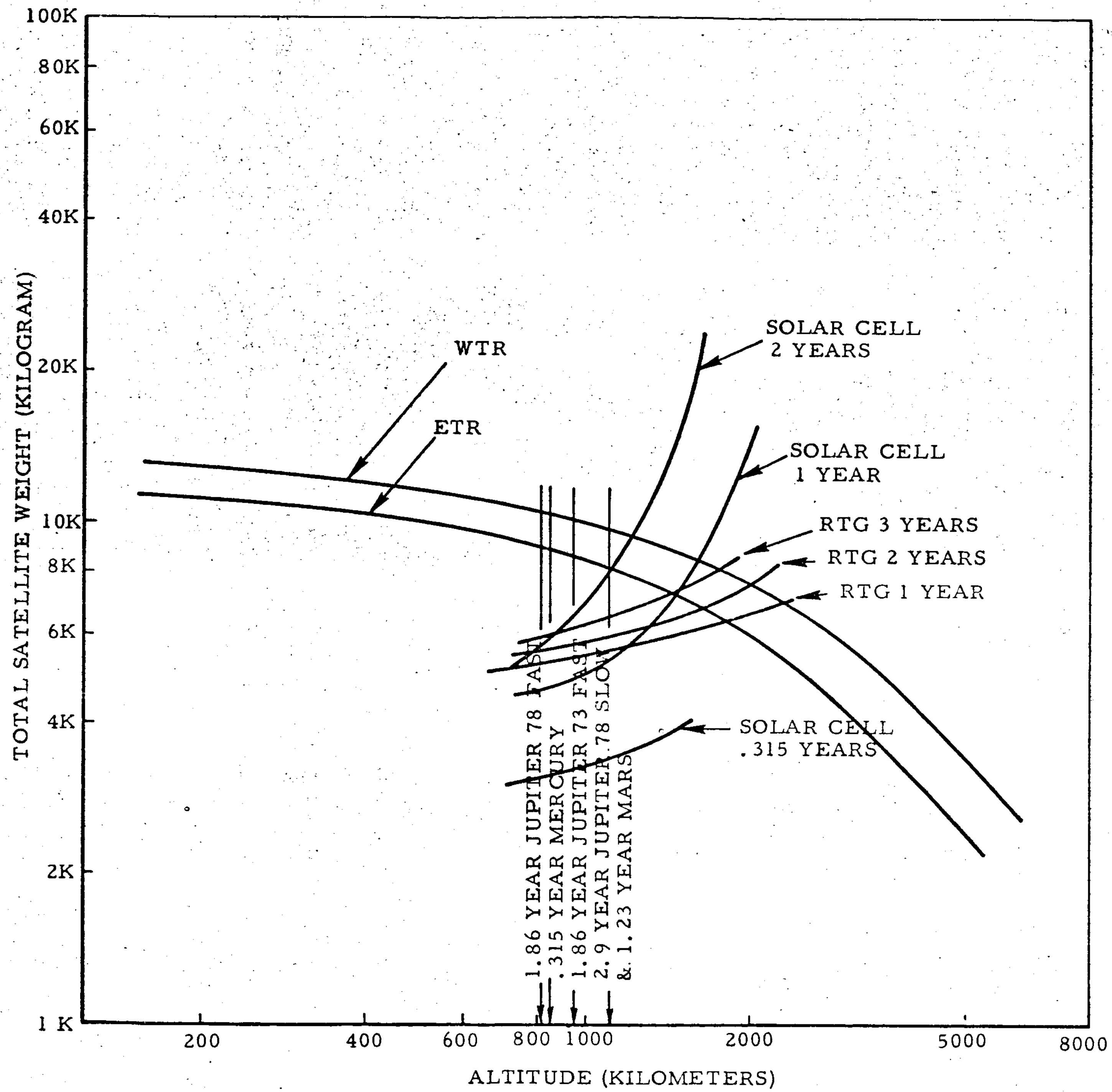


Figure 5.3.2-1. Saturn IB Satellite Weight - Profile B

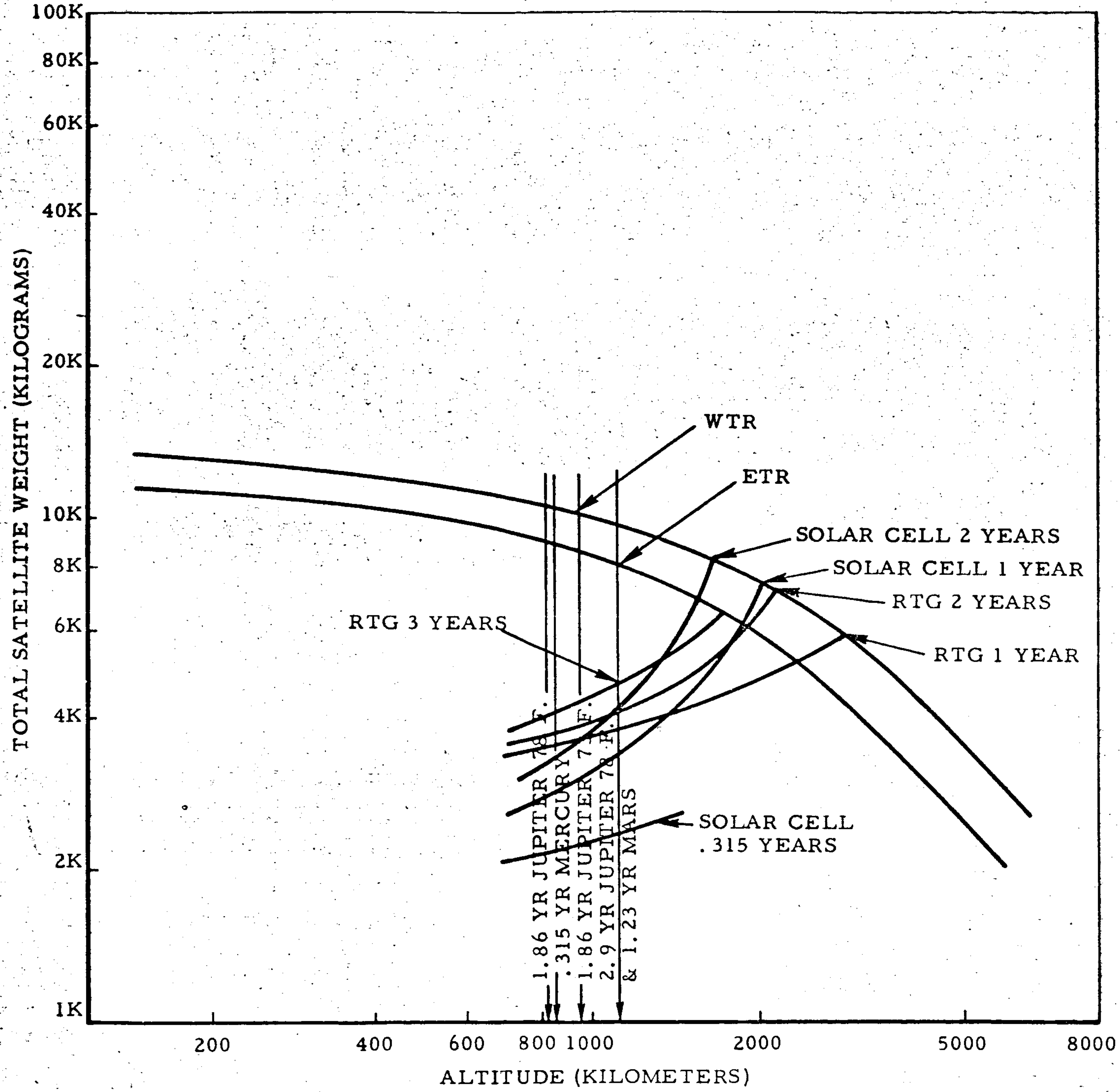


Figure 5.3.2-2. Saturn IB Satellite Weight - Power Profile E

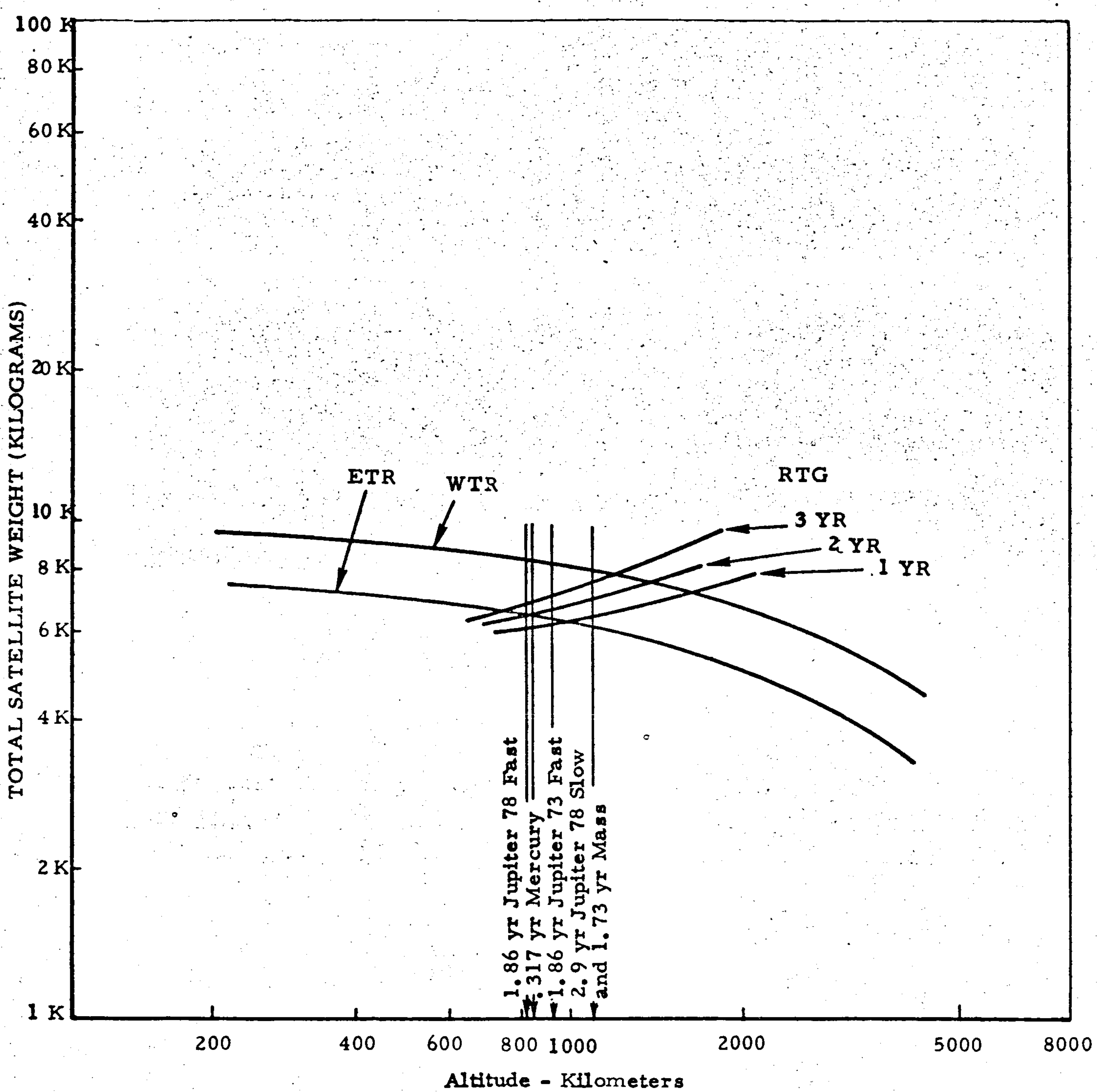


Figure 5.3.2-3. Titan IIIC Satellite Weight - Power Profile B
(Petal Antenna 6.1 Meter Diameter)

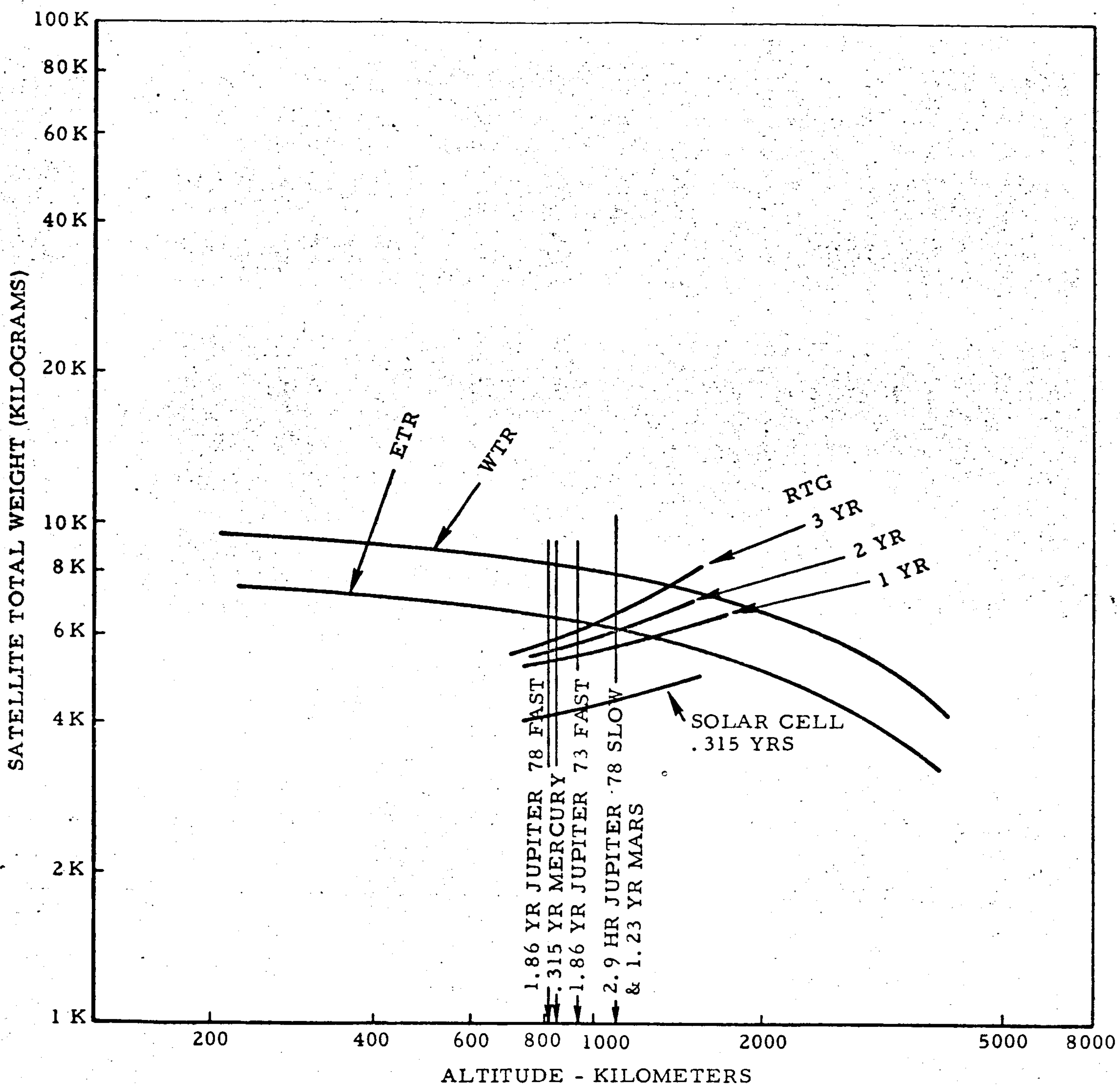


Figure 5.3.2-4. Titan IIIC Satellite Weight - Power Profile E
(Petal Antenna 9.15 Meter Diameter)

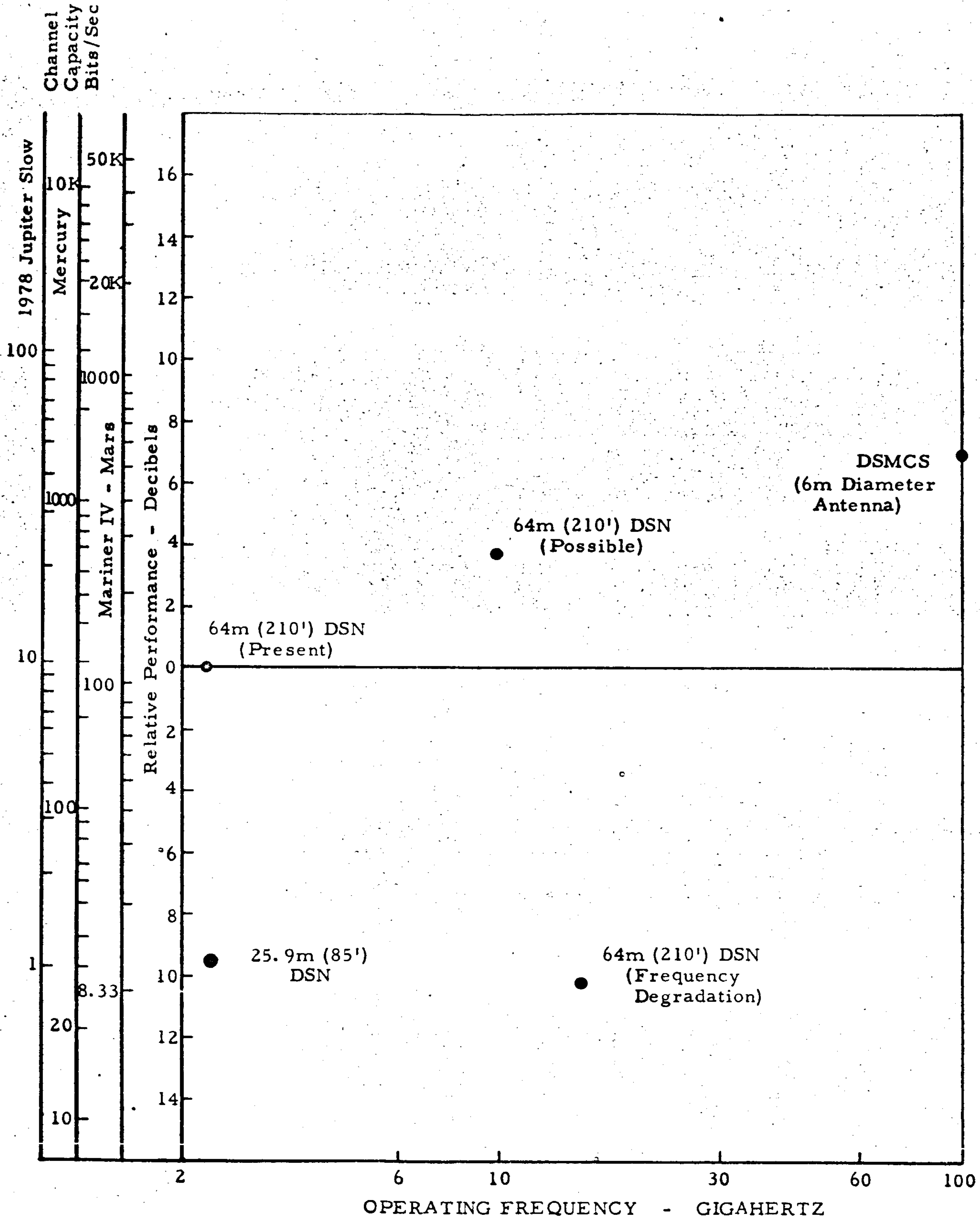


Figure 5.3.3-1. Relative Antenna Performance vs Frequency

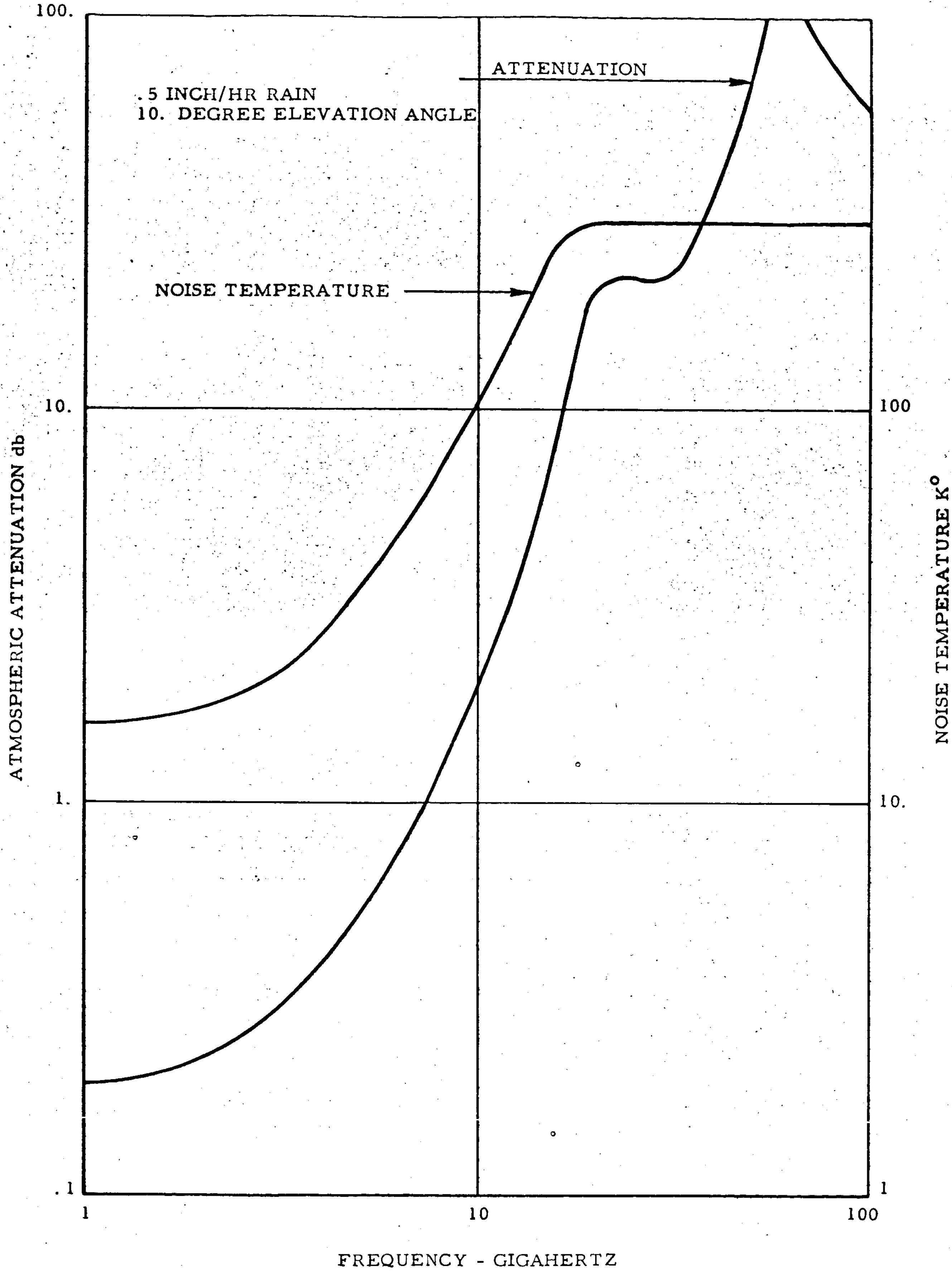


Figure 5.3.3-2. Atmospheric Attenuation vs Frequency

Table 5.3.3-2

COMMUNICATION FROM SPACECRAFT

	Present	DSMCS
Operating Frequency (Gigahertz)	2.3	100
Receive Aperture Diameter (Meters)	64 (210 ft)	6.1 (20 ft)
Receive Aperture Efficiency	.53	.5
Receive System Temperature (^{o}K)	30	40
Data Signal to Noise Power Required 5×10^{-3} Bit Error Rate (db)	7.6	7.6
Transmitter Power Output (Watts)	100	100
Data Modulation Loss (db)	4.6	4.6
Spacecraft Antenna Gain (db) 6.1 Meter Dia. (20 ft)	40.2	72.6
Performance Margin (db)	6.0	6.0
Range 1.8 AU Channel Capacity (Bits/Sec)	0.7×10^{6}	7.9×10^{6}

6.0 EXAMPLE SATELLITE CONCEPTUAL DESIGNS

Nine satellite conceptual designs were investigated considering various combinations of technologies discussed in Appendix Volume III. The designs were configured using shroud, power supply, and antenna options to provide a spectrum of representative conceptual designs. In order to assist the bookkeeping of the design options in the various satellite configurations, a descriptive nomenclature system was established to point out the primary systems associated with each configuration. The nomenclature used follows:

EXAMPLE

NB 1 S – 6 – 10^5 – 1a

Power System
- N Nuclear
- S Solar

Power Profile
- A through H

Lifetime
- Years

Launch Vehicle Shroud
- Saturn IB S
- Titan IIIC (short) T_S
- Titan IIIC (long) T_L

Antenna Aperture Diameter
- Meters

Antenna Surface Tolerance Factor
- Diameter/rms D/σ

Configuration Number and Alternate Letter

The nine configurations discussed have the following design options:

NB 1S-6-10^5-1
NB 1T_L-4-10^5-1a

SB 1 S -6-10^5-2

NB 1 T_S-10-10^4-3
NB 1 S-10-10^4-3a
NB 1 T_L-13-10^4-3b

SE 1 S-6-10^5-4
SE 1 T_S-10-10^4-4a

NE 1 T_L-4-10^5-5

The configurations will be discussed in order. Of the nine, four are preferred, they are Numbers: 1, 2, 3a and 4a. In order to compare nuclear and solar powered configurations, one year lifetimes were used for all the configurations at orbital altitudes of 1620 km.

6.1 CONFIGURATION NB1S-6-10^5-1

A line drawing of the first configuration (NB1S-6-10^5-1) is shown in Figure 6.1-1 and an isometric of the configuration is presented in Figure 6.1-2. This is one of the four preferred configurations and it is packaged in the Saturn-IB shroud. The power supply consists of three radioisotope thermoelectric generators (RTG's). Total RTG electrical output to support the B profile is 13.2 kW. This profile provides 7.5 kWe minimum full time power with 37.5 kWe available for transmitting 5 minutes per hour. Power loads consist of 30 kWe transmitter input, 6 kWe maser refrigeration and 1.5 kWe for housekeeping. In order to package the largest diameter antenna possible, the RTG's were placed on top of the satellite, a somewhat awkward position for properly supporting their mass during launch.

Communications is based on equipment operating at 100 GHz and a solid, single piece 6 meter diameter antenna with a D/σ value of 10^5 and a deployable hyperbolic sub-reflector and secondary antenna. A closed loop cryogenically cooled maser amplifier is used for receiving and the high power transmitter for command required 10 kW radiated power. Cryogenically cooled masers are considered on all the configurations with a B power profile.

The attitude control system utilizes mercury flywheels and nitrogen gas jets as noted in earlier sections of this volume. Equipment thermal loads were assumed conducted to the RTG area and radiated to space. No weight has been allocated for thermal control in any of the nuclear power configurations because of this assumption. Equipment radiation and particle protection have been accounted for in structure weight. A weight summary of the major subsystems for this configuration follows:

Power Supply	
3 RTG's	3000 kg
Battery	65 kg
Other	35 kg
	3100 kg
Communications	
Antenna	1000 kg
Maser	70 kg
Transmitter	90 kg
Refrigeration and other components	135 kg
	1295 kg
Attitude Control	270 kg
Structure and Radiation Shielding	510 kg
Total	5175 kg

6.2 CONFIGURATION NB1T$_L$-4-10^5-1a

Rearrangement of the previous configuration for the Titan was investigated resulting in configuration NB1T$_L$-4-10^5-1a. The long Titan shroud was required for equipment packaging because the short shroud did not provide adequate volume. The smaller diameter Titan shroud, however, reduced the antenna diameter to 4 meters as shown in Figure 6.2-1. The long straight sided shroud did not provide the option as did the Saturn shroud to arrange the power supply and antenna to maximize antenna diameter. The smaller diameter antenna was located at the top of the satellite and the RTG's were placed at the bottom where they are more easily supported during launch than in the previous configuration. The reduction in antenna weight, however, was offset by increased attitude control system weight required because of the increased moments of inertia imbalances in the elongated configuration. The total weight of this configuration is therefore similar to the previous configuration.

6.3 CONFIGURATION SBIS-6-10^5-2

This is the second preferred configuration and it combines an inflatable solar panel system for profile B with a 6 meter diameter antenna packaged in the Saturn-IB shroud. Conventional solar panels sized for the B profile required excessive attitude control system weight because of the extreme moments of inertia imbalances due to long solar panels. Inflatable solar panels, packaged on the top of the satellite as shown in Figure 6.3-1, were employed to reduce the moments of inertia. This was accomplished by taking advantage of the lighter weight inflatable panels and their capability of being expanded into a convenient configuration. The panel configuration used to minimize moments is shown in Figure 6.3-2 as a square deployed array. A mass was required on a boom to maintain spherical balance. The mass consisting of a battery located on a boom is shown in the figure perpendicular to the plane of the deployed solar panel. In order to maintain spherical balance, the boom and counterbalance mass rotate with the solar panel as it rotates to track the sun. Inflatable solar panel weights were assigned a conservative number of approximately 5 kg/m^2. A weight summary of this configuration follows:

Power Supply	
Inflatable Solar Panels	1760 kg
Battery	410 kg
Other Components	50 kg
	2220 kg

Communications	
Antenna	1000 kg
Maser	70 kg
Transmitter	90 kg
Refrigeration and other components	135 kg
	1295 kg
Attitude Control	1500 kg
Environment Control	
Equipment Radiator	400 kg
Structure and Radiation Shielding	600 kg
Total	6015 kg

6.4 CONFIGURATION $NB1T_S$-10-10^4-3

The short Titan shroud which is 0.6 meter larger in diameter than the long shroud was considered in this configuration. In order to increase antenna diameter significantly beyond the 4 meter diameter noted in the long Titan shroud configuration, $NB1T_L$-4-10^5-1a, a petal antenna deploying to a 10 meter diameter was considered. The stowed configuration of the antenna and location of the other major components are shown in Figure 6.4-1. This diameter shroud does not allow a high one piece center portion diameter to total diameter ratio, conducive to optimizing antenna gains. Reduced manufacturing and deployment tolerances of this diameter petal antenna when compared to one piece antenna tolerances of diameter as small as 4 meters results in approximately the same gain figure at 100 GHz. This configuration is representative of short Titan shroud configurations but is not a preferred configuration.

6.5 CONFIGURATION NB1S-10-10^4-3a

The study has pointed out gain limitations and weight penalties of petal antennas when compared to one piece antennas. The ratio of the one piece center portion or hub diameter to total diameter for petal antennas should not be overlooked, however, when comparing large diameter petal antennas with significant diameter hubs to smaller diameter one piece antennas. This point is explored in the third preferred configuration by using the large diameter Saturn-IB shroud and a 10 meter diameter petal antenna with a large hub diameter as shown in Figure 6.5-1. The deployed configuration is noted in Figure 6.5-2. Significant improvement in petal antenna performance is available in this configuration due to the relatively large hub diameter. A weight summary of the configuration follows:

Power Supply	
3 RTG's	3000 kg
Battery	65 kg
Other	35 kg
	3100 kg
Communications	
Antenna	3000 kg
Maser	70 kg
Transmitter	90 kg
Refrigeration and other Components	135 kg
	3295 kg
Attitude Control	300 kg
Structure and Radiation Shielding	900 kg
Total	7595 kg

6.6 CONFIGURATION $NB1T_L$-13-10^4-3b

This configuration is found in Figure 6.6-1 and the long Titan shroud was used to package a shroud limited petal antenna. The maximum petal antenna diameter available in this shroud was 13 meters with other equipment packaged as shown in the figure. This configuration was an exercise in packaging petal antennas in long shrouds. The low ratio of the center hub diameter to total antenna diameter does not optimize antenna gain and the remainder of the configuration is similar to configurations previously discussed.

6.7 CONFIGURATION SE1S-6-10^5-4

Power profile B did not permit use of conventional solar panels primarily because of excessive attitude control system weight requirements associated with moments of inertia of the long panels.

A primary power load influencing the area of the solar panel was the continuous 6 kWe load in profile B for maser refrigeration. Reduction in this continuous load dramatically reduced solar panel area requirements and permitted the use of conventional solar panels. This configuration took advantage of a reduction in this load as found in power profile E, where only 1 kWe was assigned for partial refrigeration of a parametric amplifier rather than complete refrigeration of a maser. Solar panel projected area was reduced from 360 m^2 to 170 m^2. This still resulted in large attitude control system weights but not excessive as would have been the case with power profile B and conventional solar panels.

The configuration shown in Figure 6.7-1 is packaged in the Saturn-IB shroud with the maximum diameter one piece antenna of 6 meters. Configuration SB1S-6-10^5-2, noted in Section 6.3, required one boom and counter weight designed to follow perpendicular to the inflatable solar panel to maintain spherical balance. This configuration required two stationary booms and counter weights to maintain spherical balance.

6.8 CONFIGURATION SE1T_S-10-10^4-4a

This is the fourth preferred configuration. It takes advantage of the attitude control system weight savings of power profile E. The short Tital shroud is used for packaging a 10 meter diameter petal antenna. The relationships of the major components are shown in Figure 6.8-1. Figure 6.8-2 illustrates the deployed solar panels and the two stationary booms and counter weights in an isometric view. A summary of the weights follows:

Power Supply

Solar Panels	1240 kg
Battery	200 kg
Other Components	25 kg
	1465 kg

Communications	
Antenna	2800 kg
Parametric Amplifier	25 kg
Transmitter	90 kg
Refrigeration and other Components	100 kg
	3015 kg
Attitude Control	1950 kg
Environment Control	
Equipment Radiator	400 kg
Structure and Radiation Shielding	1025 kg
Total	7855 kg

This total weight averages 1500 kg in excess of Titan IIIC ETR payload capability for the missions studied. Launches from WTR would provide adequate payload for all missions except for the Mercury flyby.

6.9 $NE1T_L$-4-10^5-5

The SNAP-10B power supply could offer considerable capability in the mid 70's. The large volume of the SNAP-10B's however did not permit their use in the selected shrouds for the B power profile. The long Titan shroud and power profile E would permit packaging four SNAP-10B's as shown in Figure 6.9-1. The four power units would be hinged at the equipment compartment and deployed away from their clustered position shown in the figure after orbit injection. The configuration considered a 4 meter diameter solid antenna with a fixed hyperbolic sub-reflector.

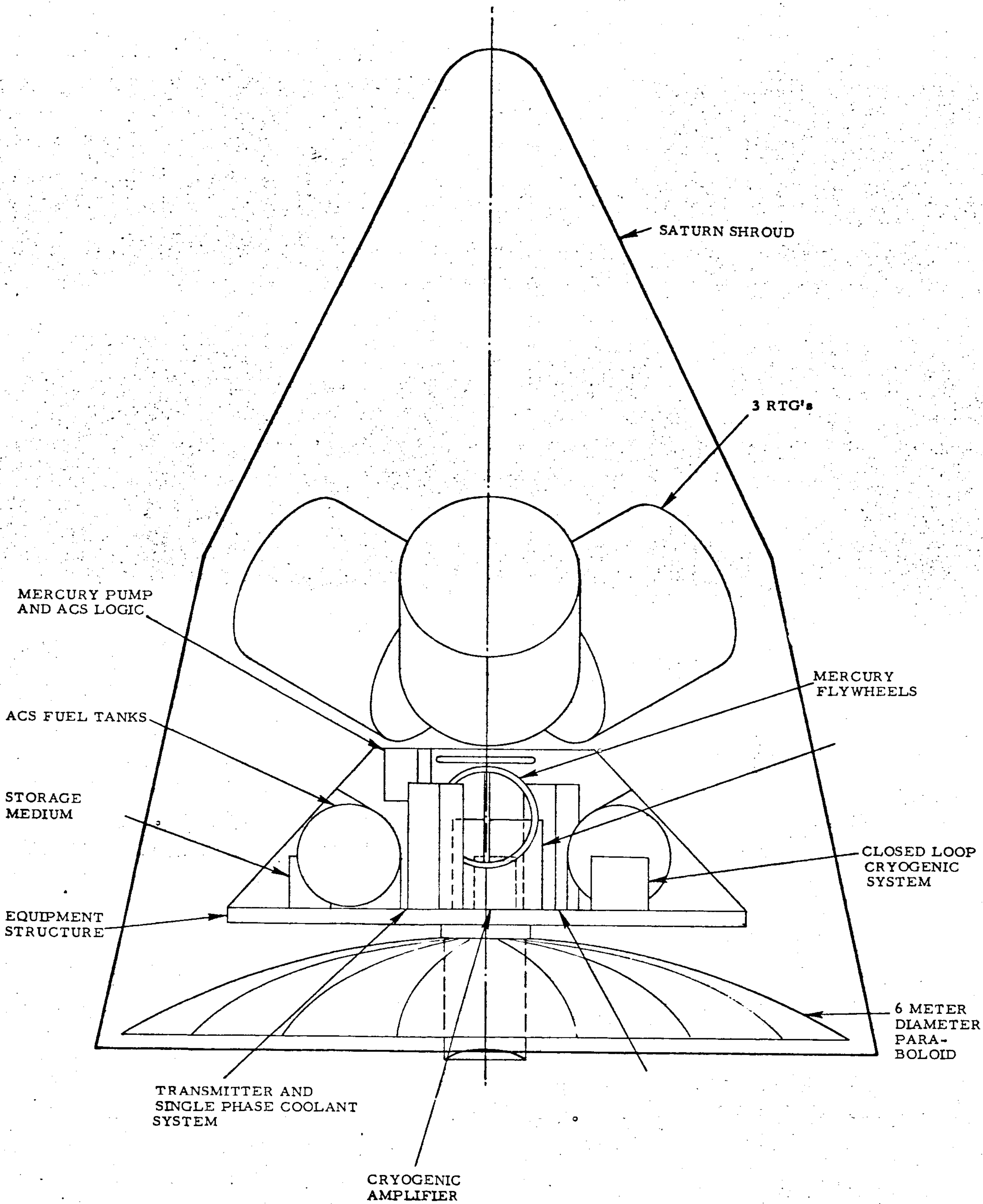


Figure 6.1-1. DSMCS Configuration NBIS-6-10^5-1

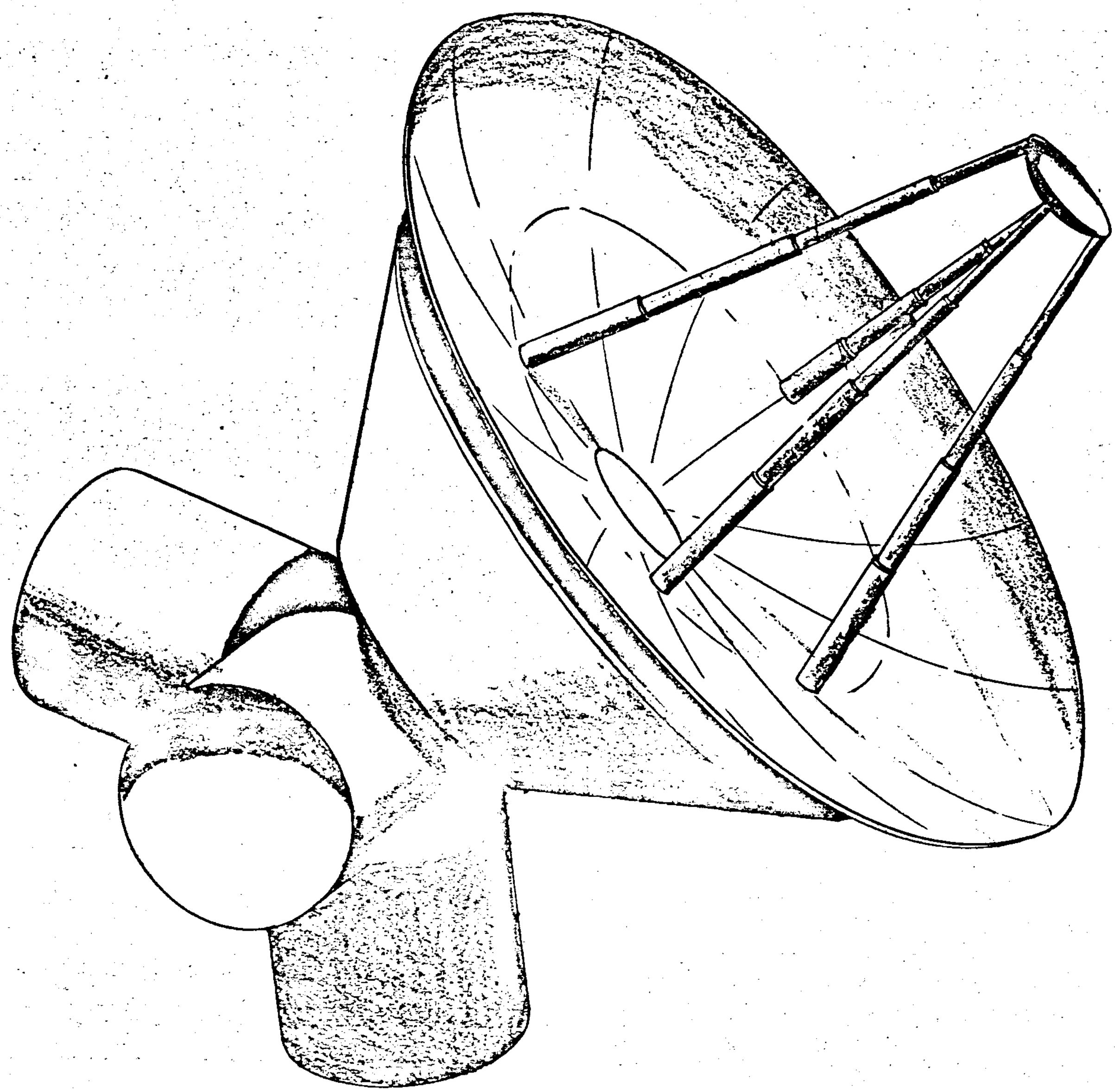


Figure 6.1-2. DSMCS Configuration NBIS-6-10^5-1

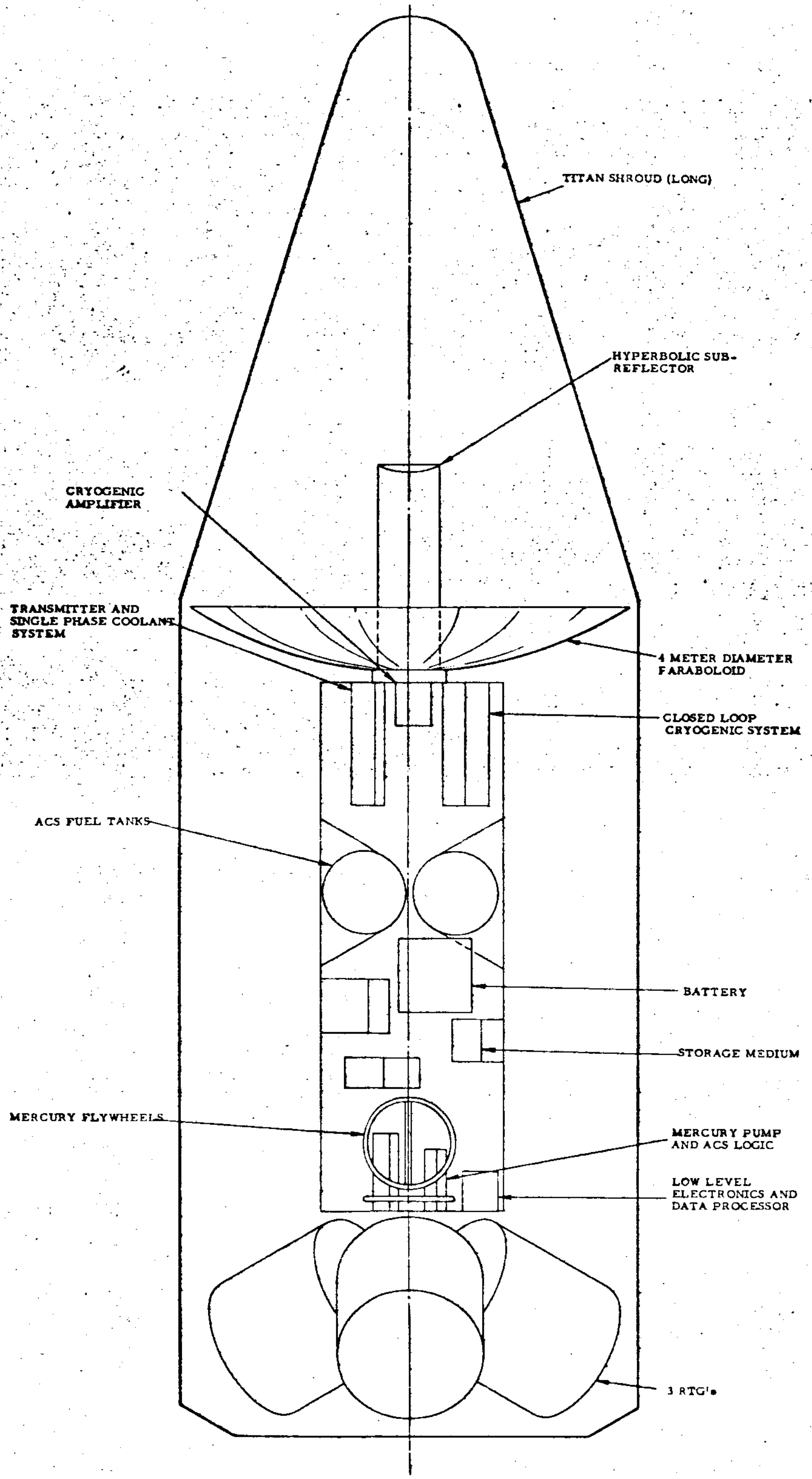


Figure 6.2-1. DSMCS Configuration $NB1T_L$ -4-10^5-1a

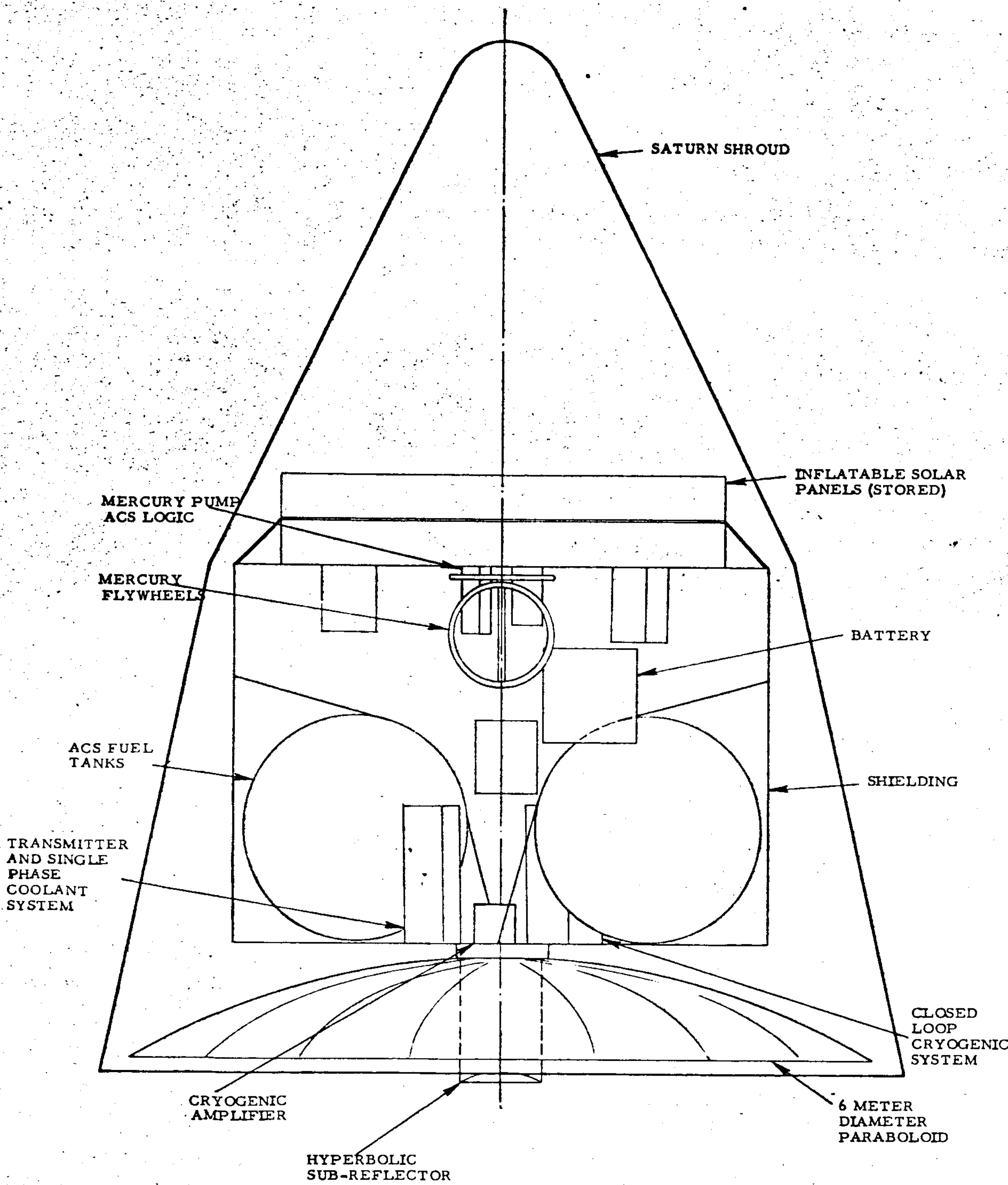


Figure 6.3-1. DSMCS Configuration SBIS-6-10^5-2

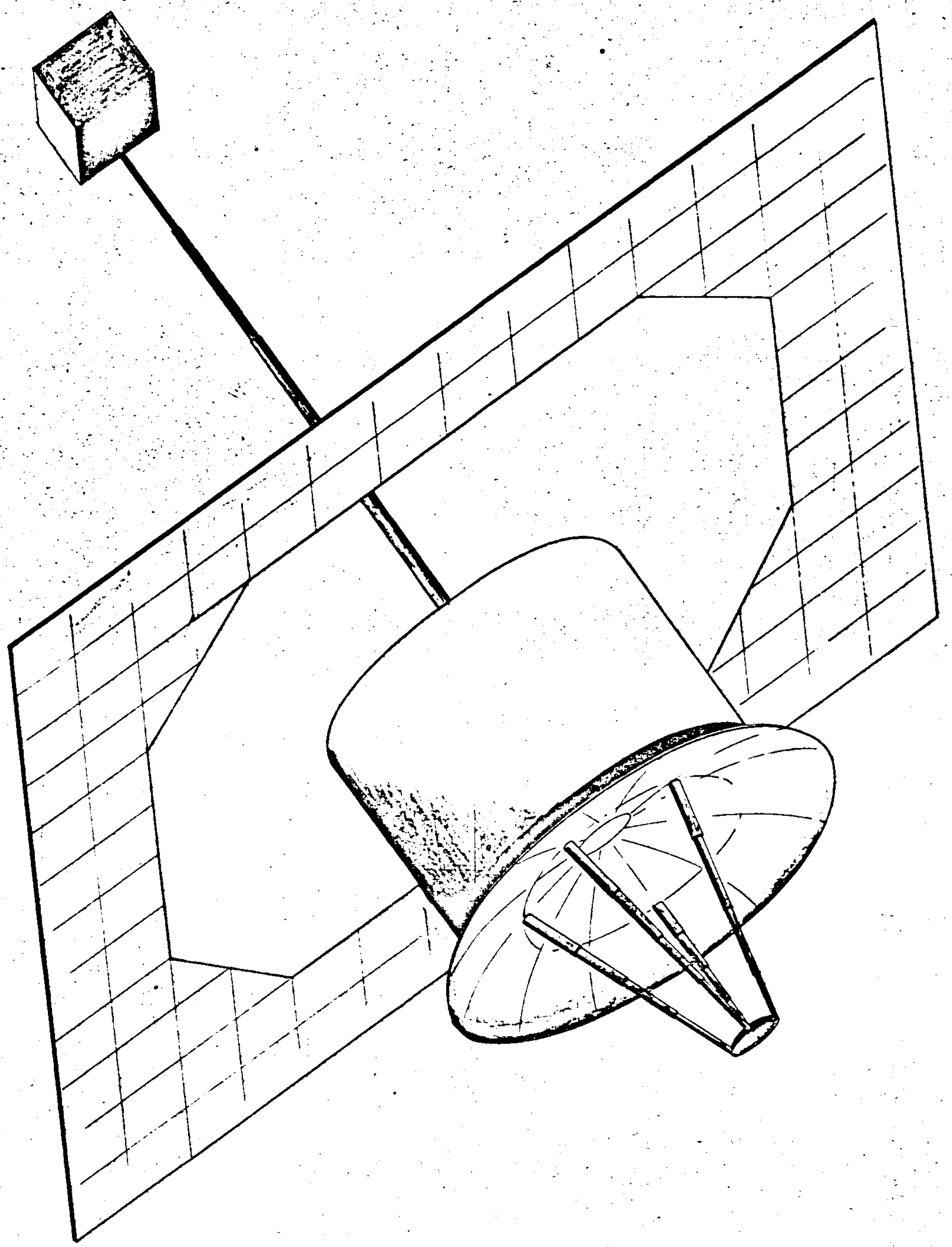


Figure 6.3-2. DSMCS Configuration SBIS-6-10^5-2

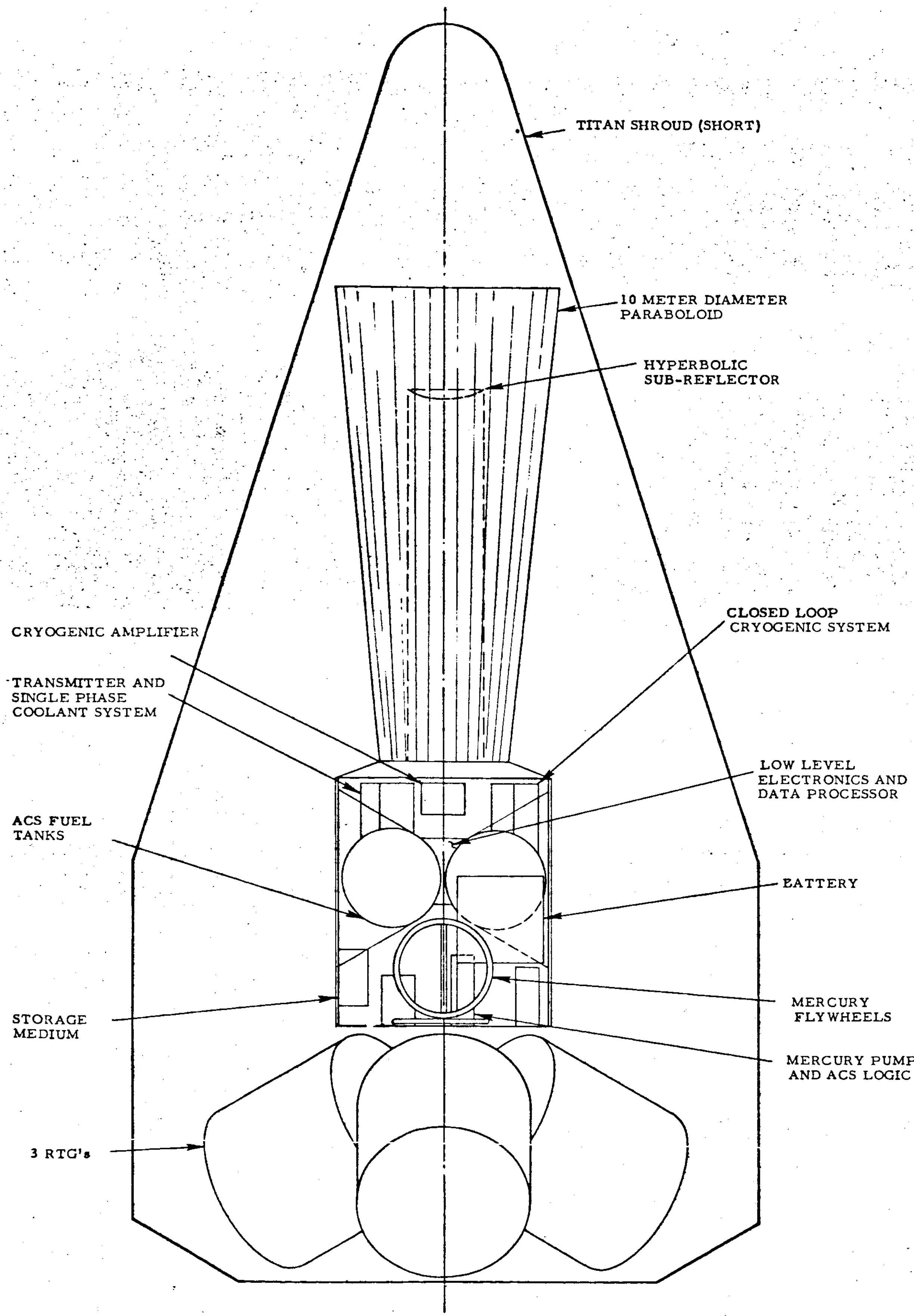


Figure 6.4-1. DSMCS Configuration NB1T_S-10-10^4-3

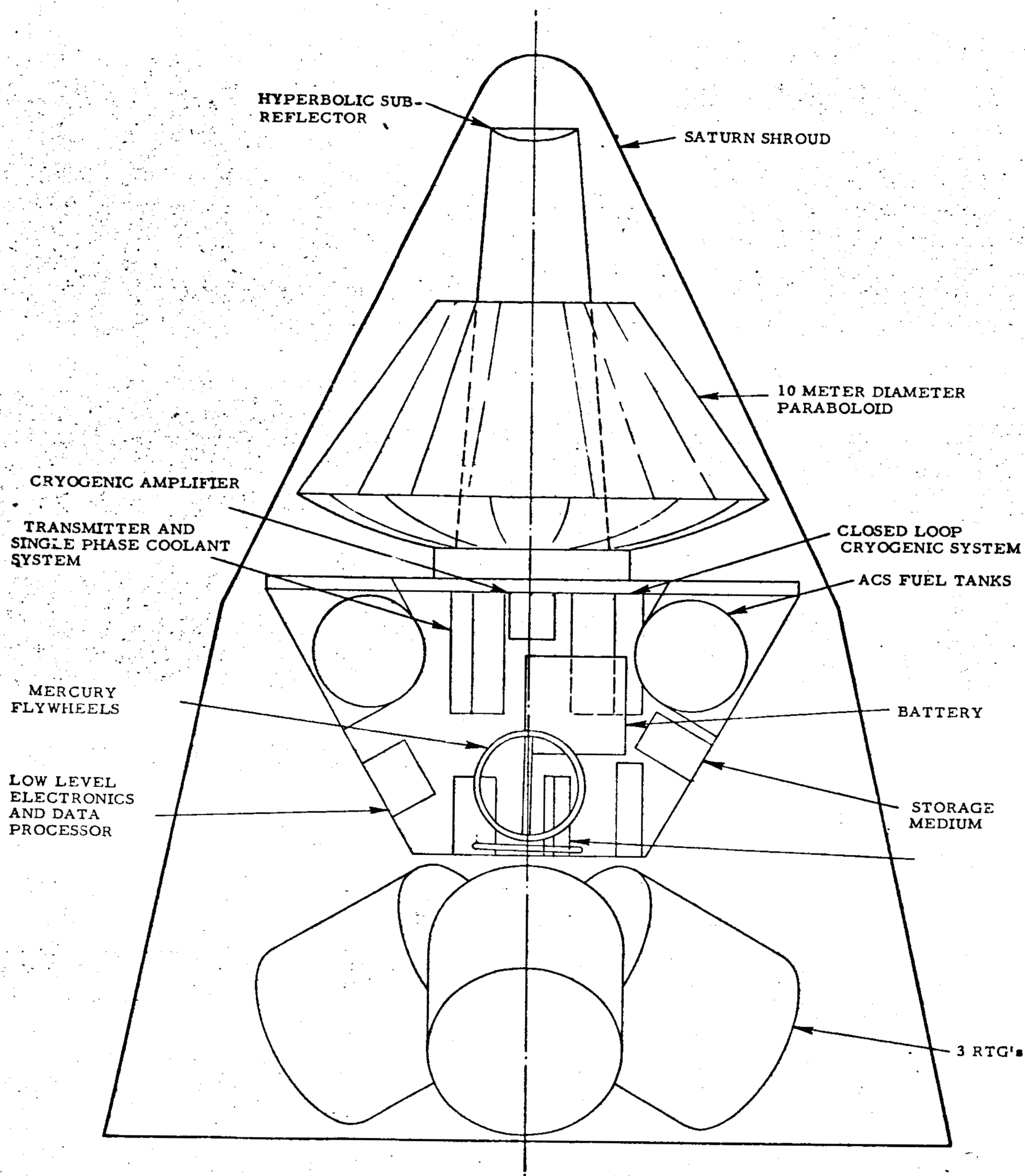


Figure 6.5-1. DSMCS Configuration NBIS-10-10^4-3a

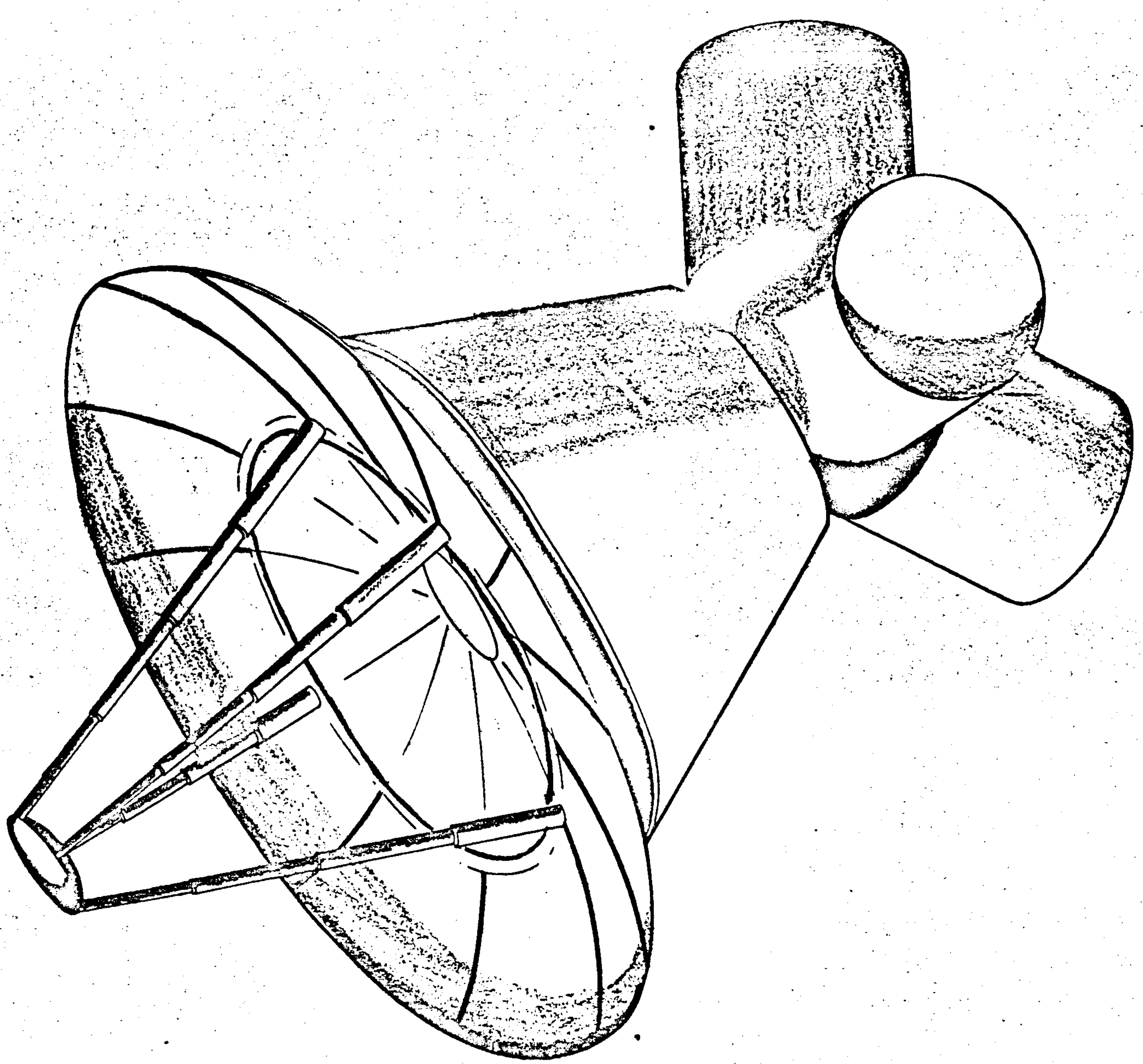


Figure 6.5-2. DSMCS Configuration NBIS-10-10^4-3a

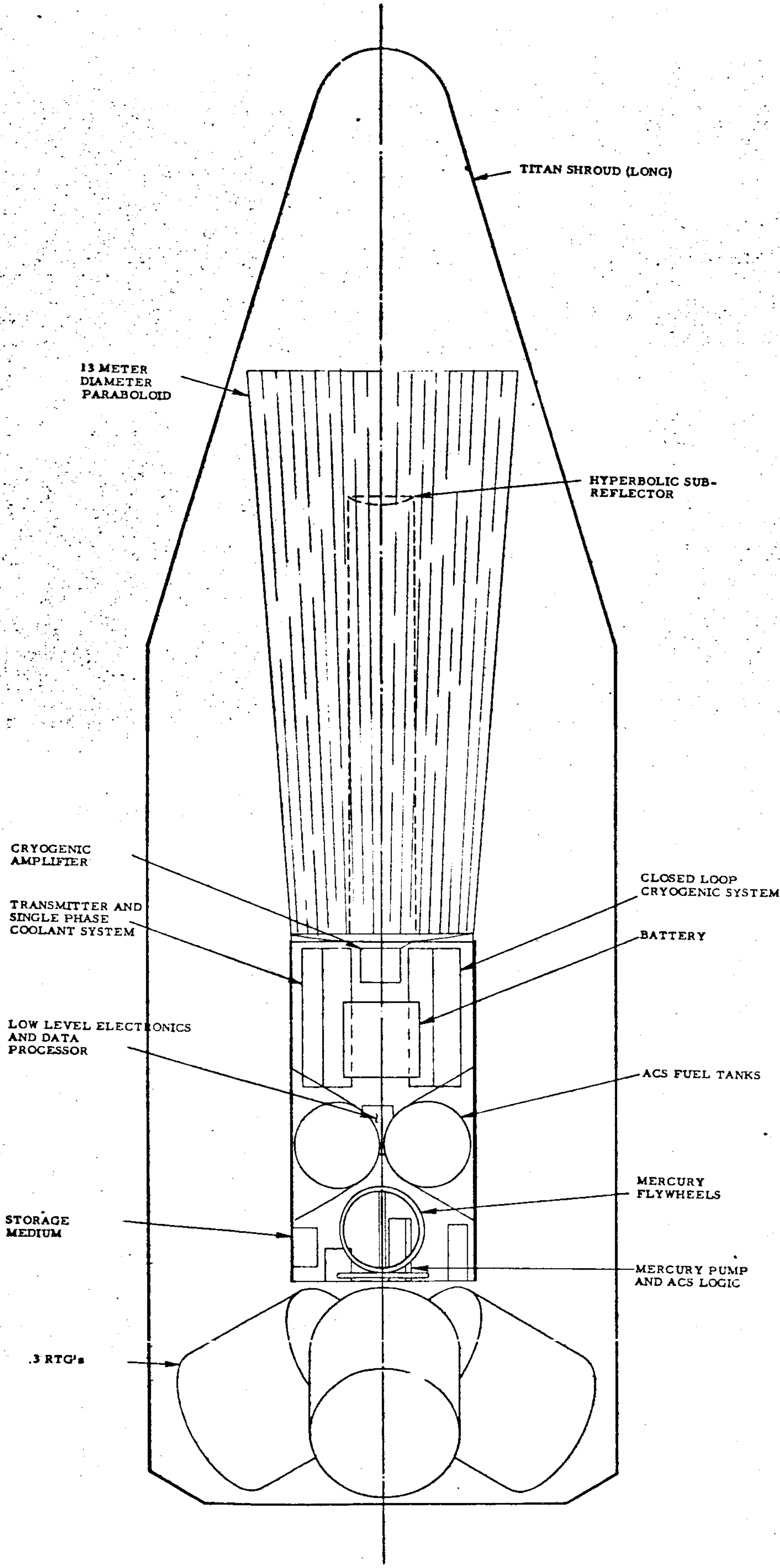


Figure 6.6-1. DSMCS Configuration $NB1T_L$-13-10^4-3b

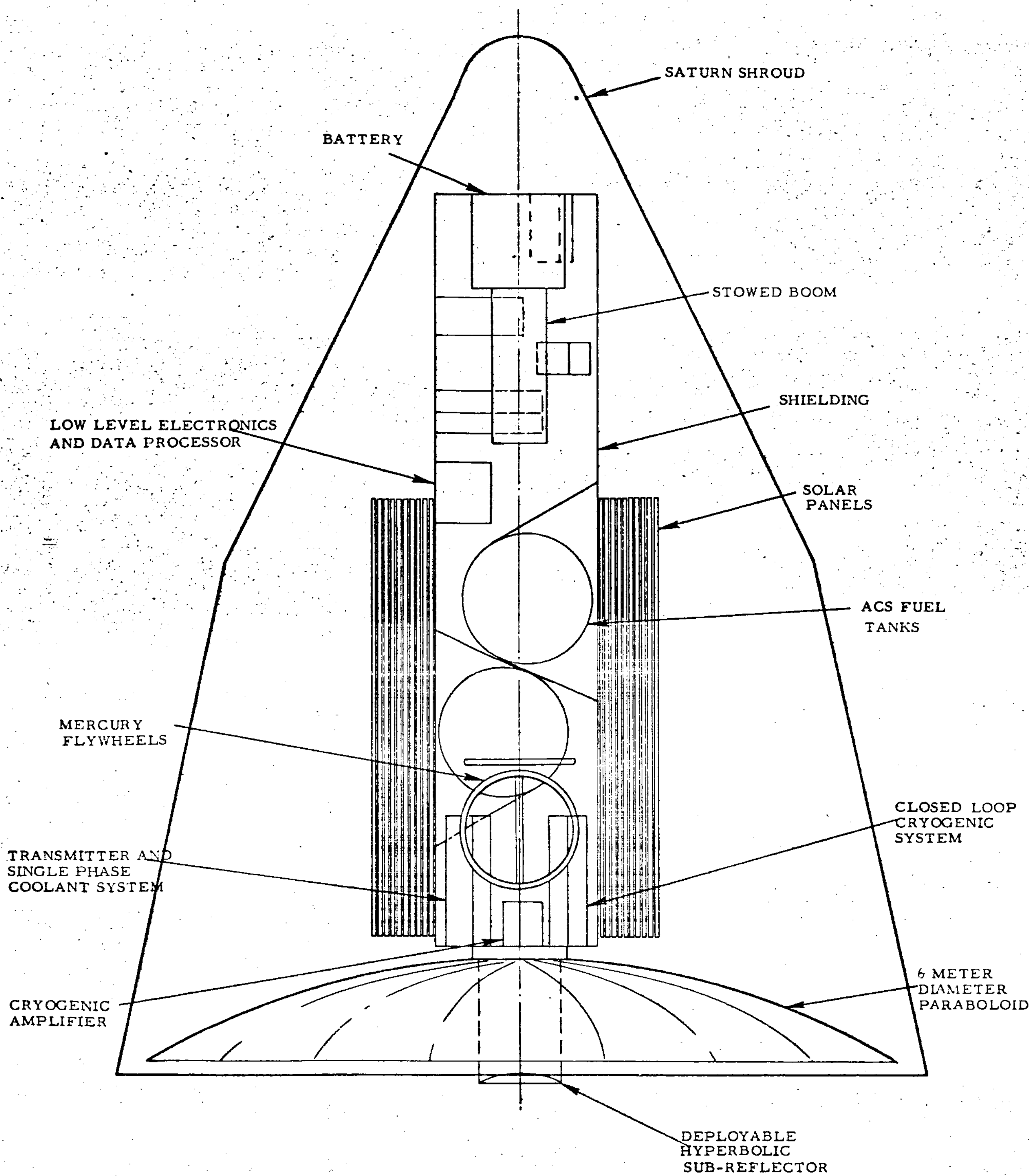


Figure 6.7-1. DSMCS Configuration SEIS-6-10^5-4

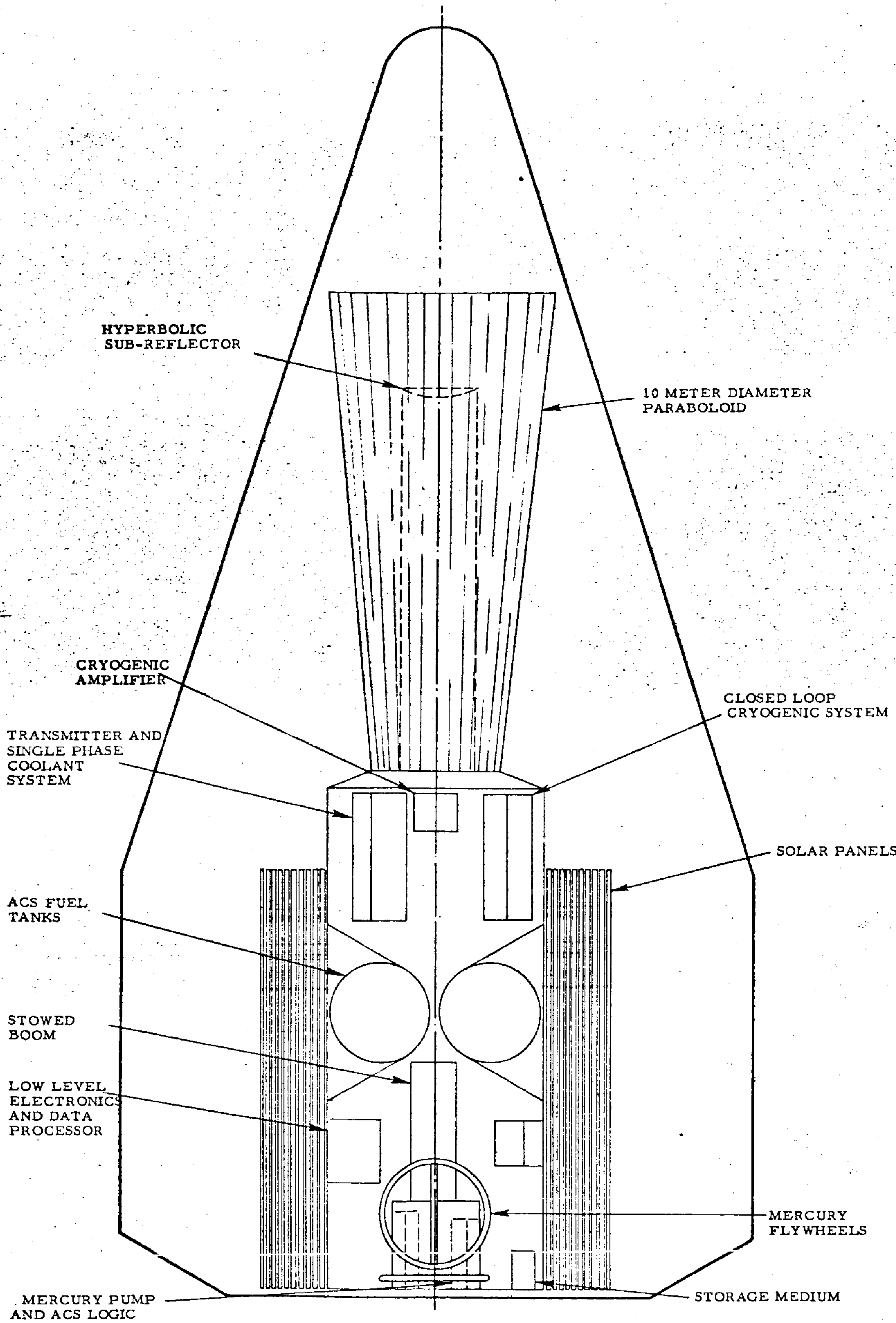


Figure 6.8-1. DSMCS Configuration $SEIT_S$-10-10^4-4a

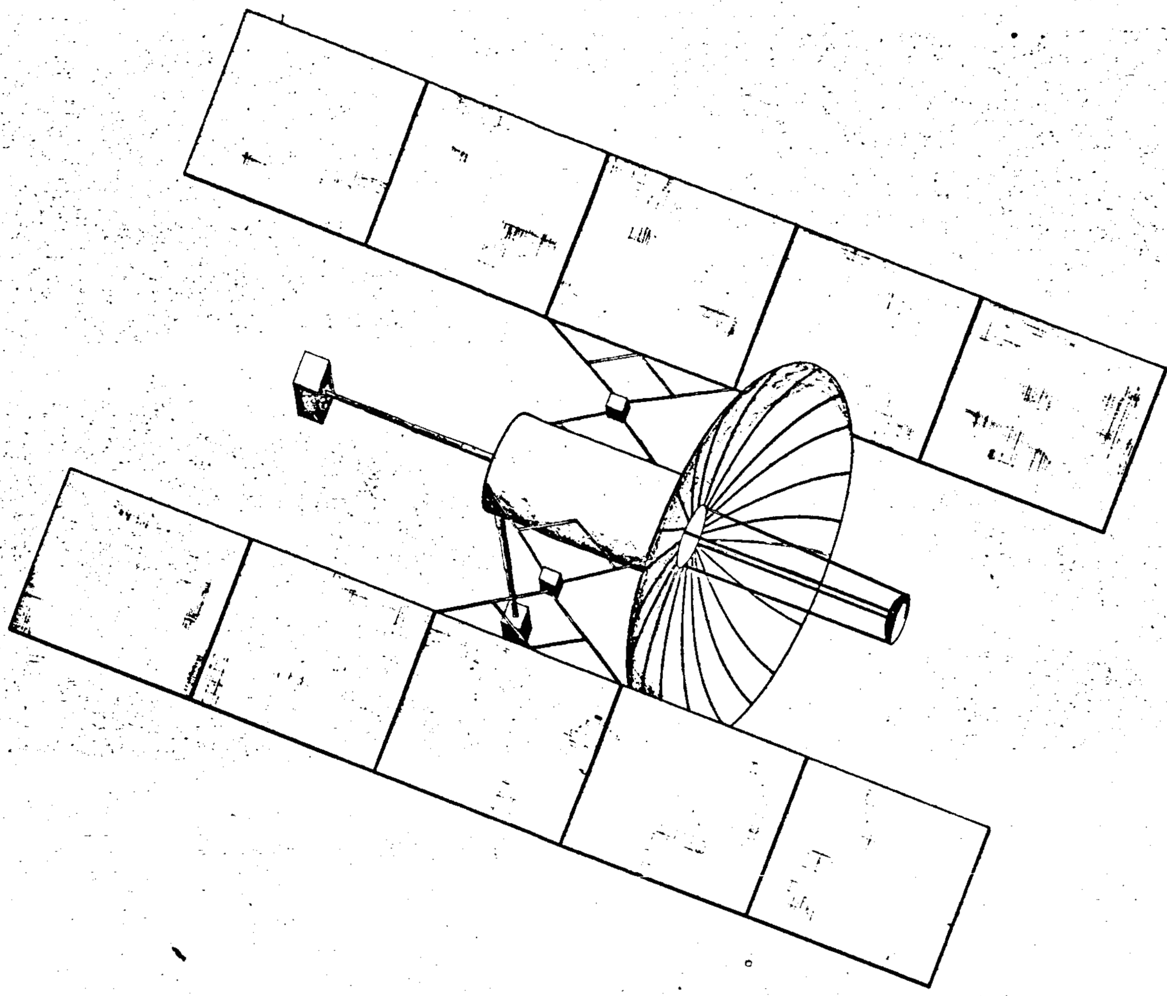


Figure 6.8-2. DSMCS Configuration $SEIT_S$-10-10^4-4a

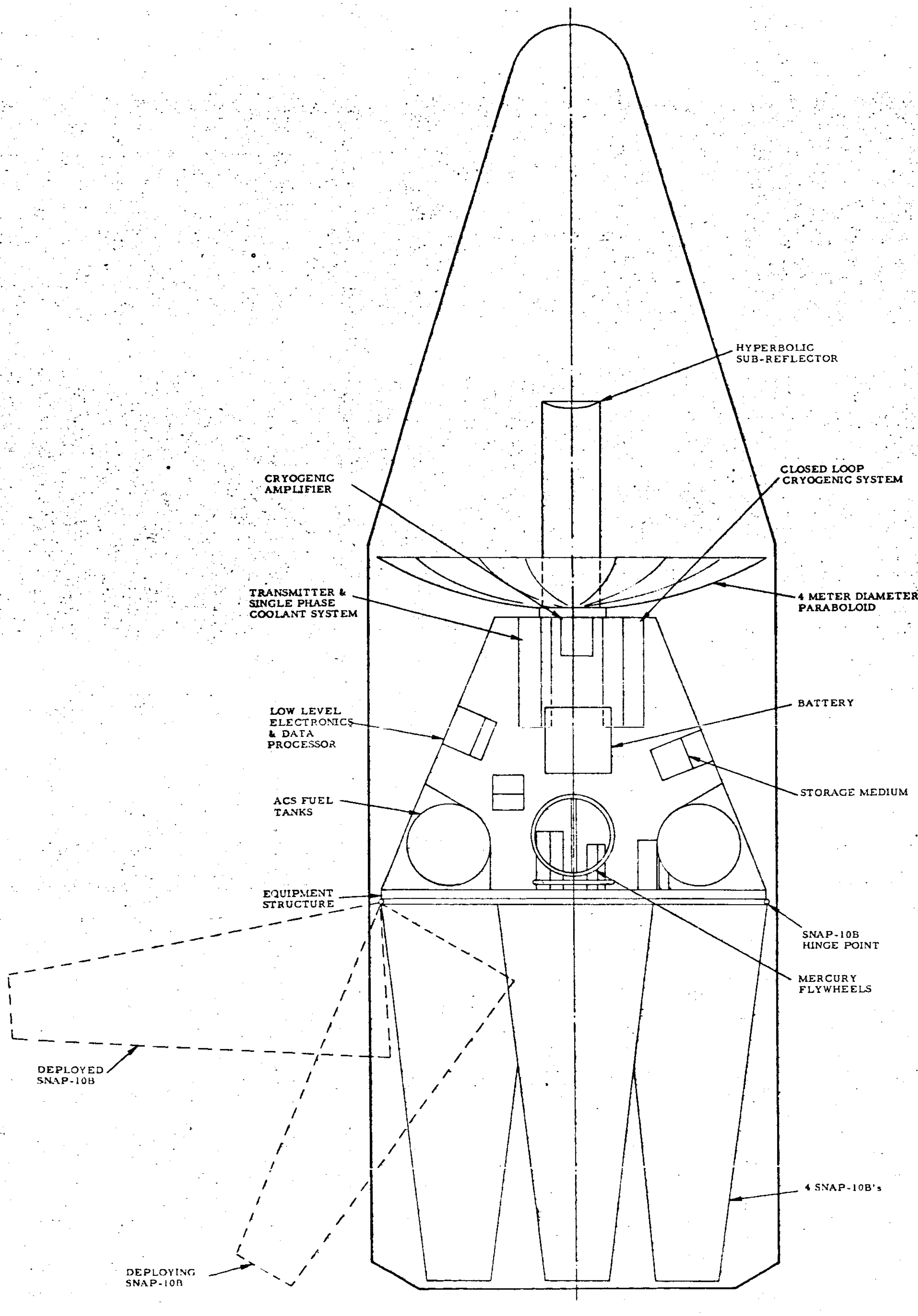


Figure 6.9-1. DSMCS Configuration $NEIT_L$-4-10^5-5

7.0 FEASIBILITY, CAPABILITIES AND TECHNOLOGICAL PROBLEM AREAS

The study results have pointed out that the DSMCS concept would be feasible in the 1975-1985 period only if major technological areas are developed to a degree well beyond the present state of their art. The list of technological problem areas found later in this section attests to this. Establishing the above degree of technical feasibility was performed, however, without consideration to development cost or cost effectiveness.

The DSMCS operating at 100 gigahertz would provide over 6 db of improvement for receiving data from deep space spacecraft compared to present S-band earth based systems. In addition, the DSMCS would provide continuous monitoring of spacecraft without antenna switching, require only one spacecraft acquisition and reduce ground network complexity. Use of frequencies above S-band permits higher antenna gains both on the satellite and spacecraft with present antenna technology. Orbiting a DSMCS for each spacecraft mission would relieve ground station loading. A more flexible and less costly operating procedure could take advantage of manned assistance on the DSMCS for time sharing the receiver and monitoring many spacecraft.

Engineering and technology problems have been discussed throughout the study. Engineering problems have been defined as problems requiring extension to or improvement of known state of the art. Two examples of engineering problems are given for example. Stellar sensors required for the attitude control system were found not to have reliability in keeping with the 3-year maximum DSMCS lifetime. The engineering solution proposed either improvement in stellar sensor reliability or redundant units or a combination of both. This is a proposed engineering solution as compared to a technological solution requiring breakthroughs in stellar sensor technology. A second engineering example concerns erection and operation tolerance of the antenna feed. This problem does not necessarily require a technological advance but rather an exploitation of high accuracy structural erection engineering procedures.

The many engineering problems discovered during the study were handled in the above manner. There were problems, however, that fell beyond the definition of engineering and required substantial development or research in their respective technologies. These areas are listed below as technological problem areas that require research and solution before the DSMCS concept will be fully implemented.

Technological Problem Areas

- 100 GHz cryogenic amplifiers
- Closed loop cryogenic coolers
- 100 GHz efficient transmitters
- Surface characteristics and deployment tolerances of spaceborne antennas larger than 6 meters diameter (Saturn-IB solid dish shroud limit)
- RTG unit power levels
- Increase SNAP-10A output/weight to proposed SNAP-10B capabilities
- Radiation resistant solar cells
- Lightweight inflatable solar panels

Engineering Problem Areas

- DSMCS and Spacecraft attitude control, receiver, oscillator and transmitter improvements

8.0 LOGISTICS, SCIENCE AND MANNED APPLICATIONS SUMMARY

8.1 ESTIMATE OF SPACECRAFT TRAFFIC VS CALENDAR DATES

There are numerous mission time tables available that provide tentative flight schedules during the period of interest of DSMCS. Unmanned missions that could be supported by DSMCS are presented in Table 8.1-1 through 1980. Manned planetary missions are also capable of being supported by DSMCS. Their flight schedule however, is more difficult to predict. This study selected one typical manned Mars launch opportunity in the early 1980's as an example for DSMCS investigations. A manned Venus mission was investigated early in the study but it did not satisfy study ground rules and therefore was not persued in detail.

8.2 RESUPPLY FOR SATELLITE LIFETIME EXTENSION

Power supply lifetime at DSMCS orbital altitudes were of primary concern in selecting appropriate systems. Radiation damage to solar cells limited their utility to satellite configurations with only one year lifetime. Increase in panel area to provide longer lifetimes increased attitude control fuel weight requirements beyond practical limits. Solar panel systems therefore were not competitive when compared to long lifetime RTG systems with reliability figures projected to the period of DSMCS operations. Resupply of degraded solar panels could improve satellite utility by extending lifetime. Degraded panels would be removed during satellite rendezvous with a manned vehicle and replaced with new panels.

In addition to solar panel resupply, attitude control fuel resupply could extend satellite lifetime. Satellite configurations with solar panels required large attitude control fuel supplies. These configurations would particularly benefit from fuel resupply and this capability would change the panel design philosophy. Larger panels increasing satellite performance could be considered without limiting satellite lifetime.

8.3 SATELLITE UTILITY FOR SCIENCE INSTRUMENTATION

8.3.1 RADIO ASTRONOMY

Radio astronomy telescopes operating above the earth's atmosphere offer greater capabilities than a comparable earth-bound system constrained by atmospheric windows. Adopting DSMCS to radio astronomy will open large portions of the spectrum previously unavailable for study. Planetary and cosmic source spectrum between 2 cm and 10 microns wavelength are obscured by atmospheric absorption. Advances in predetection amplifiers and detectors in the lower decade of this range have produced environment limited radio astronomy systems rather than detector limited systems even in the few windows in this range. The smoothing enhancement gained by tracking with ground based antennas is invalidated by the variable atmospheric insertion loss. A radio astronomy system using DSMCS as a platform could not only be made much more sensitive than a ground system but could monitor planetary or cosmic activities continuously. Above the earth's atmosphere, cooler planets such as Jupiter and Saturn may be studied with greater sensitivity and in more detail. The detectability is increased not only by the increased radiometer sensitivity for a comparable earth receiver but also by the f^2 power density relation of thermal bodies.

DSMCS orbits were selected to provide continuous tracking of planetary spacecraft. During the latter portion of the spacecraft trajectory, the DSMCS would also provide continuous monitoring of the target planet. Cosmic events could be monitored during the entire spacecraft transfer. The pointing accuracy and stability of the DSMCS would permit useful radio astronomy experiments.

8.3.2 OPTICAL ASTRONOMY

The Astronomy Working Group at the Iowa Summer Study Group 1962 was continued at the Space Science Board's Woods Hole, Massachusetts meeting in 1965. Optical astronomy in space was defined to include all astronomical research carried out with reflecting telescopes in space at wavelengths from 800 Å to 1 mm, excluding solar studies. The Working Group on Optical Astronomy considered the possibilities for studying stars, star systems, nebulae, and planets from orbiting telescopes sensitive to electromagnetic radiation at wavelengths between 800 Å and 1 mm. The group made the following recommendations that are related to DSMCS capabilities:

1. Two or more telescopes having apertures of 40 inches or larger should be included in the Apollo Extension Systems (AES) program. The Orbiting Astronomical Observatory program should be continued until AES launchings are definitely scheduled.

2. Development of various detectors required in space telescopes should be supported by NASA.

3. Development of improved gratings would be of central importance in the space astronomy program.

4. Development of optical interferometers should be pressed, with probable initial operation on the ground.

5. Research and development concerned with problems of space telescope optics, especially with the primary mirror, should be supported by NASA.

The group also concluded that a space telescope of large diameter, with a resolution corresponding to an aperture of at least 120 inches, detecting radiation between 800 Å and 1 mm, is becoming technically feasible and will be uniquely important to the solution of the central astronomical problems of our era. The group recommended that this telescope because of the long lead time and large cost be manned to assure reliability.

This recommendation infers that smaller telescopes using the DSMCS as an unmanned platform could be useful to the scientific community. In addition, a DSMCS with the large 120 inch diameter aperture, coupled to a manned module would provide a combined satellite reducing attitude control and power requirements in the manned module.

8.4 MANNED SATELLITE APPLICATIONS

Unmanned DSMCS's provide a stable platform with high gain communications capability and with power levels capable of supporting manned satellite requirements. Manned satellites with limited lifetimes coupled to DSMCS's in low altitude orbits could extend lifetimes of the manned satellites and increase their operational scope. The manned operational scopes could be increased to perform communications relay functions and manned planetary spacecraft return guidance assistance. Long duration orbital biological experiments could also be performed. Earth environment experiments in addition to the radio and optical astronomy experiments noted in the previous section could be crew operated on the DSMCS/manned module combined satellite.

Solar power configured DSMCS's would require only minor redesign to accommodated docking and fuel line and power line connections. Nuclear power configured satellites, however, would require RTG shielding redesign to increase radiation protection due to the proximity of the manned module.

Future manned earth orbiter interests could use the DSMCS as a building block to perform more sophisticated operations for extended periods of time. Utilizing the DSMCS as a staging area introduces interesting applications. Removing the continuous line of sight between the DSMCS and the planetary spacecraft constraint, permits exploitation of DSMCS capabilities for manned operations considering reduced radiation shielding and increased payload capabilities. The DSMCS staging area could be used for power and communication assistance during rendezvous and assembly of manned modules. The assistance would be applicable to large assembled manned earth orbiters or to modular manned planetary spacecraft. The DSMCS could provide logistics functions during module assembly.

Table 8.1-1

NASA ADVANCED UNMANNED MISSIONS (TYPICAL)

Mission	Type	Weight (kilograms)	Number of Launches 68	69	70	71	72	73	74	75	76	77	78	79	80	Total
Voyager-Mars	Orbiter/Capsule	2700-3600						2		2		2				6
Advanced Voyager-Mars	Orbiter/Lander	18-27,000										2		2		4
Voyager-Venus	Orbiter/Capsule	2700-3600							2		2		2			6
Mariner-Jupiter	Flyby	350-450							2		2					4
Voyager-Jupiter	Orbiter	1100									2		2			4
Mariner-Mercury	Flyby	225-300						1	1	1	1					4
Mariner-Mercury	Capsule	500									1	1	1	1		4
Pioneer-Asteroid	Fly-thru	180			1	1		1								3
Mariner-Asteroids	Eros Flyby	225-275					1		1							2
Mariner-Asteroids	Vesta-Flyby	350				1		1		1						3
Mariner-Comets	Tail Fly-Thru	250							1			1			1	3
Advanced Pioneer	Interplanetary	200		2	2	1	1	1	1	1	1					10
Pioneer	Out-of-the Ecliptic (20^{o})	65				1	1	1	1	1						5
Pioneer	Out-of-the Ecliptic (high angles)	90									1	1	1	1	1	5
Solar Monitors	Interplanetary	50-75				1	1	1	1	1	1	1	1	1	1	10